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PRELIMINARY
APOLLO SPACECRAFT REQUIREMENTS SPECIFICATION

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30 September 1962 (Revised)
15 September 1962

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PRELIMINARY
APOLLO SPACECRAFT REQUIREMENTS
SPECIFICATION
U

30 September 1962 (Revised)
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NAS9-150

Approved by

J. W. Paup
Vice President and Apollo Program Manager

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NORTH AMERICAN AVIATION, INC.
SPACE and INFORMATION SYSTEMS DIVISION

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APOLLO SPACECRAFT REQUIREMENTS SPECIFICATION

1. SCOPE

1.1 Scope. - This specification defines the following aspects of the Apollo Spacecraft:

(a) Composition and performance requirements of the Command Module (C/M) and Service Module (S/M)

(b) Performance characteristics of the major systems and modules

(c) Integration and interfacing of the major systems and modules within the Spacecraft

Although the landing of men on the surface of the Moon, and their safe return to Earth is the ultimate objective of this Project, the design approaches and operational techniques to transport a 3-man crew to the vicinity of the Moon are defined herein. The Lunar Excursion Module, although a part of the Spacecraft, is not included in this specification. Future revisions shall define the discrete LEM contributions.

1.2 Configuration. - The final form of the Spacecraft utilized in the Apollo program shall be a function of the particular mission involved. The Spacecraft configuration shall be dictated by the Lunar Landing Mission requirements.

1.3 Specification Organization. - The Apollo General Requirements are contained within three separate specifications as follows:

Apollo Mission Requirements, SID 62-700-1
Apollo Spacecraft Requirements SID 62-700-2
Apollo Ground Operations Requirements SID 62-700-3

1.3.1 Specification Relationship. - The Apollo specifications relationship is shown on Figure 1.

2. APPLICABLE DOCUMENTS

2.1 Applicability. - The following documents of the issue in effect on the date of the contract, form a part of this specification to the extent specified herein.

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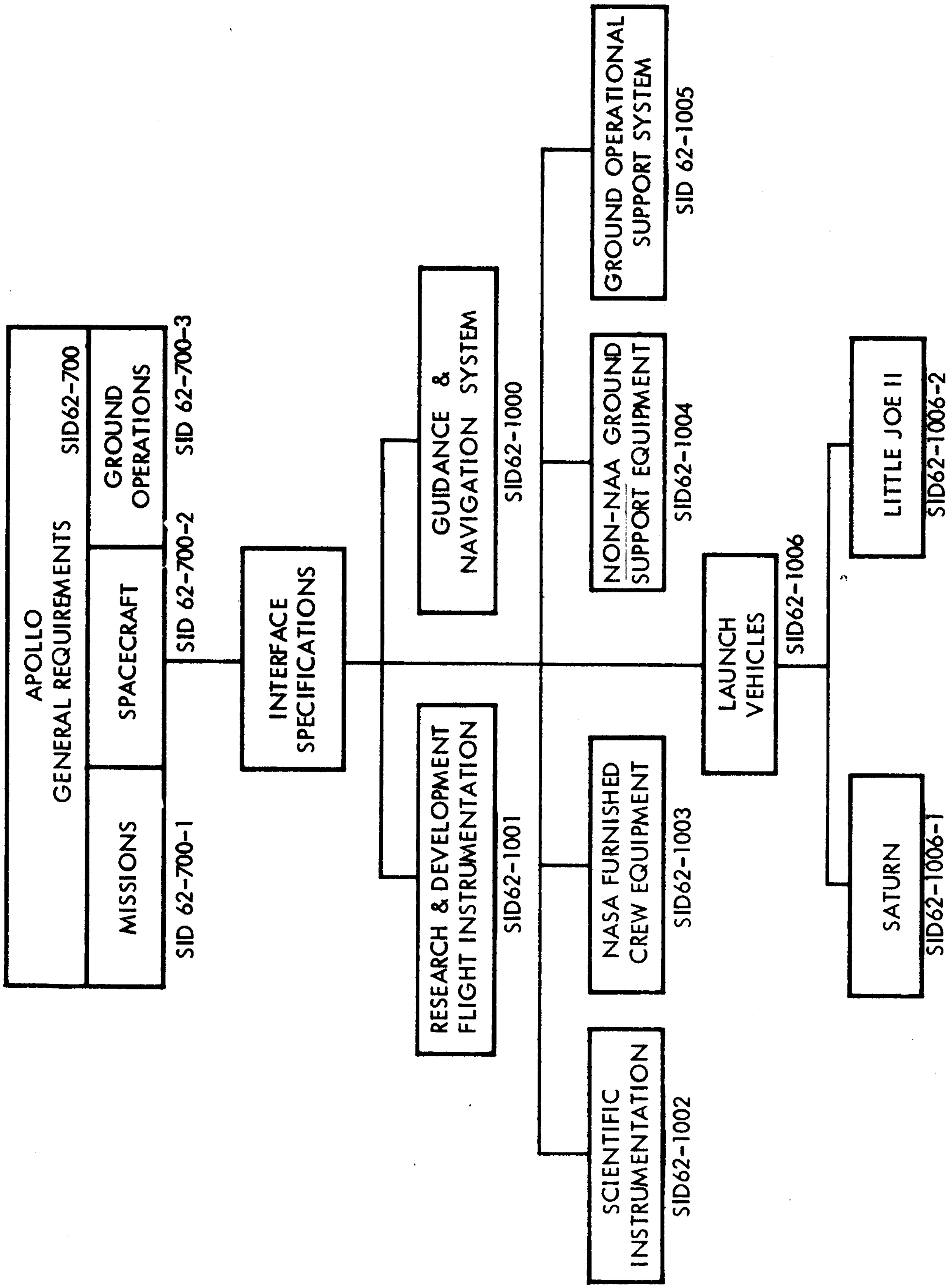

Figure 1. Apollo Specification Relationship Type I

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SPECIFICATIONS

North American Aviation, Inc.

Space and Information Systems Division

SID 62-700-1	Apollo Mission Requirements Specification
SID 62-700-3	Apollo Ground Operations Requirements Specification
SID 62-1000	Guidance and Navigation System Interface Requirements Specification
SID 62-1001	R & D Flight Instrumentation Interface Requirements Specification
SID 62-1002	Scientific Instrumentation Interface Requirements Specification
SID 62-1003	NASA Furnished Crew Equipment Interface Requirements Specification
SID 62-1004	Non-NAA Ground Support Equipment Interface Requirements Specification
SID 62-1005	Ground Operational Support System Interface Requirements Specification
SID 62-1006-1	Saturn Launch Vehicle Interface Requirements Specification
SID 62-1006-2	Little Joe II Launch Vehicle Interface Requirements Specification
MC 999-0002	Electromagnetic Interference Control for the Apollo Space System

STANDARDS

Military

MIL-STD-130	Identification Marking of U.S. Military Property
MS 33586	Metals, Definitions of Dissimilar

CONFI[redacted]

PUBLICATIONS

National Aeronautics and Space Administration (NASA)

NCP-200-2	Quality Program Provisions for Space System Contractors

Military Handbook

MIL-HDBK-5	Strength of Metal Aircraft Elements

DRAWINGS

Marshall Space Flight Center

10M01071	Environmental protection when using electrical equipment within the areas of Saturn complexes where hazardous areas exist, procedure for.

[redacted]

2.2 Precedence. - The order of precedence in case of conflict shall be as follows:

(a) The contract

(b) This specification

(c) Other documents referenced herein.

3. REQUIREMENTS

The contents of this specification represents the basic design characteristics of the Apollo Command and Service Modules and shall be used as a design goal in establishing the overall Spacecraft operational and functional characteristics. The Command Module, Service Module and adapter combination shall be designed with consideration for additional modules which will accommodate a Lunar Landing Mission. The environmental and analytical design shall be oriented to include the ultimate Lunar Landing Mission concept.

3.1 Characteristics. - The following paragraphs delineate the characteristics and requirements for the Command Module and Service Module.

3.2 Spacecraft. -

3.2.1 Geometric Parameters. -

3.2.1.1 Dimensions. -

Overall height	644.5 $\pm$1.6 inches
Diameter	154 $\pm$0.5 inches
Structure Outline	See Figure 2

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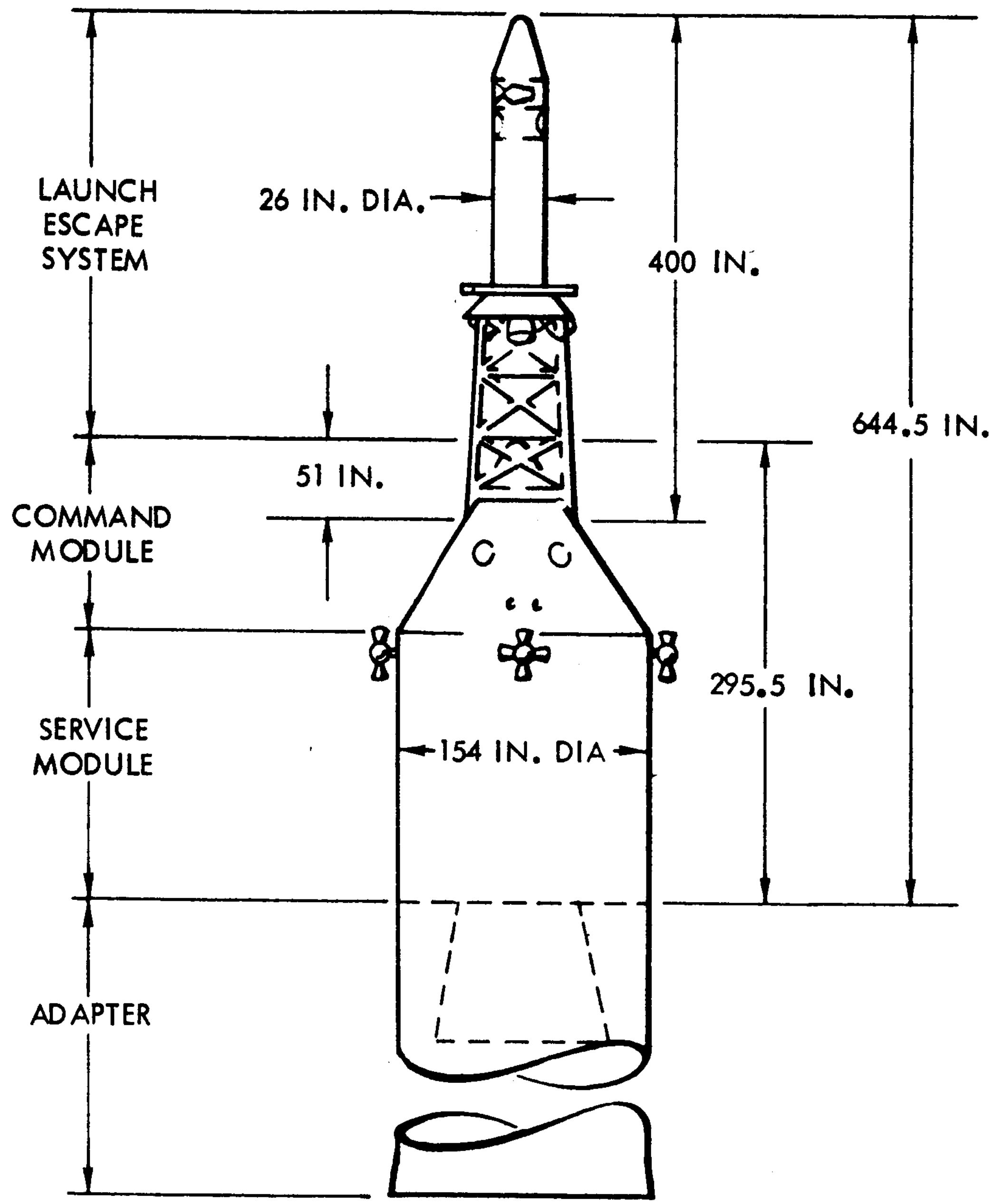

Figure 2. Apollo Spacecraft

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3.2.1.2 Design Objective Weight. -

Gross injected weight = 90,000 pounds

3.2.2 Boost Stabilization. - Windage, aerodynamic effects, variations of the center of gravity, etc. shall be compensated for by the launch vehicle during the boost phase.

3.2.3 Performance. - The following paragraphs summarize the nominal performance capabilities of the Command Module and Service Module.

3.2.3.1 Communications. - During all mission operations (except on the far side of the moon), ground communications with and tracking of the Command Module and Service Module will be accomplished through the GOSS network. The exact stations in communication with the Command Module and Service Module for any particular mission phase are stipulated in the GOSS Performance and Interface Specification SID 62-1005.

3.2.3.2 Trajectories. - The general Command Module and Service Module trajectories are detailed in the Apollo Mission Specification (SID 62-700-1). After translunar injection, the primary measured Spacecraft positional accuracy shall be provided by the Guidance and Navigation system.

3.2.3.3 Propulsion. - Propulsion increments of the Command Module and Service Module shall be satisfied by the Service Propulsion System and shall be utilized as indicated during the following mission phases.

Mission Phase	Incremental Velocity (fps)
Translunar Midcourse	310
Lunar Orbit Injection	3785
Lunar Maneuvers	Undetermined
Transearth Injection	4103
Transearth Midcourse	310

Propulsion increments involved with the ascent, injection into parking orbit and translunar injection shall be supplied by a NASA furnished launch

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vehicle. The launch vehicle third stage shall mate directly with the Spacecraft adapter and shall be an SIV B or a compatible interface equivalent.

3.2.3.4 Aerodynamics. - The aerodynamic properties of the Spacecraft during ascent shall be compatible with stable launch dynamics. The Spacecraft (Command Module only) aerodynamics shall be capable of vehicle transfer from the entry interface to the recovery interface with maneuverability compatible with the mission and crew tolerances.

3.2.3.5 Attitude Control. - From lift off through translunar injection, attitude control shall be supplied by a system external to the Spacecraft. During the remainder of the mission the Spacecraft modules shall fulfill the attitude requirements as defined in paragraph 3.3.6 (Stabilization and Control System).

3.2.3.6 Guidance and Navigation. - Refer to paragraph 3.4.1.3.1.6 for the performance requirements of the Guidance and Navigation System.

3.2.4 Launch Vehicle Requirements. - The Spacecraft shall be injected into an earth parking orbit by a launch vehicle capable of achieving a 90 to 400 nautical mile altitude orbit (within safe boost trajectory limits) dependent upon the pre-programmed mission profile.

3.2.5 Interfaces. - Refer to paragraph 3.5.

3.2.6 Environments. - The Spacecraft environments shall be as defined in paragraph 3.10 of this specification.

3.3 Major Systems. - The Command Module and Service Module shall consist of the following major systems:

1. Communication and Instrumentation
2. Electrical Power
3. Environmental Control
4. In-Flight Test
5. Structural

6. Stabilization and Control

7. Reaction Control

8. Crew

9. Launch Escape

10. Earth Landing

11. Guidance and Navigation

12. Service Propulsion

Systems 1 through 6 are discussed in the following paragraphs (3.3.1 through 3.3.6 respectively).

The Reaction Control System (RCS), System 7, is composed of independent systems in the Command Module and Service Module under the control of the Stabilization and Control System. The Reaction Control System requirements shall be defined in the Command Module and Service Module sections, as applicable.

The requirements for systems 8 through 11 are defined in the Command Module section paragraph 3.4.1.

The Service Propulsion System, system 12, is entirely contained within the Service Module and the requirements are contained in paragraph 3.4.2.

3.3.1 Communication and Instrumentation System (C&I). -

3.3.1.1 Requirements. - The communication and instrumentation system shall be designed to provide the following capabilities:

(a) Convert to electrical signals those physical parameters which must be displayed, recorded, or transmitted
(b) Obtain photographic records of events occurring inside and outside the command module.

(c) Provide an optical means for observing events external to the Spacecraft with magnification as required to give adequate detail

(d) Convert optical data to electrical signals through the use of TV equipment

(e) Condition electrical signals to a common level to allow convenient display, recording and transmission

(f) Provide stable frequencies for synchronization of time-dependent Spacecraft subsystems except for the guidance and navigation equipment, and provide a time reference for all time-dependent Spacecraft operations

(g) Assemble and encode, in suitable form for transmission, the data required by the Mission Control Center during Spacecraft missions

(h) Store for future readout that data which cannot be transmitted to the ground in real time

(i) Provide for voice communication between crew members, and provide for the transfer of audio signals to and from transmitters and receivers

(j) Provide antennas to transmit and receive electromagnetic radiation in a directable manner during near earth phases

(k) Provide one antenna to transmit and receive electromagnetic radiation in a directable manner during deep space phases

(l) Transmit voice, telemetry, and TV signals to the GOSS stations

(m) Receive and demodulate voice transmissions from the GOSS stations

(n) Receive and transmit in phase coherence a Deep Space Instrumentation Facility (DSIF)-type signal to enable the GOSS stations to track the Spacecraft in angle, velocity, and range during deep space phases

(o) Receive and respond to interrogating radar signals for tracking purposes during near earth phases

(p) Provide electronic recovery aids for location and recovery of the spacecraft following entry

(q) Display to the crew the information necessary for control of the Spacecraft and for determining mission performance

(r) Provide signals for display to the crew of the output of the TV cameras

3.3.1.2 Equipment. -

3.3.1.2.1 RF Electronic Equipment Group. -

3.3.1.2.1.1 VHF FM Transmitter. - This transmitter shall have a minimum output of 10 watts and shall operate in the 225-260 mc band to transmit telemetry data during near-earth phases. The transmitter output shall be coupled to the VHF Broad Band Antenna via the multiplexer. Spare components shall be provided within the spacecraft as needed.

3.3.1.2.1.2 VHF AM Transceiver. - The VHF AM transceiver shall be used for voice communications between Spacecraft and earth during near earth phases of lunar missions, between Spacecraft and belt-pack transceiver and between Spacecraft and rescue aircrafts after earth landing. In addition, it shall be used as a back up beacon during recovery in case of main beacon failure. The transmitter shall have a minimum output of 10 watts, operating AM, or CW in the 243-300 mc band and utilizing the VHF Broad Band Antenna during near earth and the VHF Recovery Antenna during Recovery and Post-landing operations, via the multiplexer.

3.3.1.2.1.3 DSIF Transponder Equipment. - The DSIF Transponder Equipment shall be phase coherent transmitting and receiving equipment which provides two-way voice communications, telemetry and television transmission from the Spacecraft, coherent two-way doppler tracking, and turn around PRN ranging during deep space operation. During earth orbital missions, it shall be used to transmit real time television.

(a) DSIF Receiver-Transmitter - The DSIF Receiver shall be used for receiving voice or ranging signals from the earth station. It shall receive PM and FM/PM signals at 2110-2120 mc through the 2-kmc High Gain Antenna or the 2-kmc Omni Antenna and the diplexer. It is a double conversion superheterodyne type receiver incorporating a voltage controlled oscillator phase locked to the received frequency. A second voltage controlled oscillator is included for non-coherent operation and generation of FM signals. The DSIF Transmitter is used to transmit telemeter data, real time television, analog voice and ranging signals. It shall have a minimum output of 200 mw and operated PM or FM at 2200-2400 mc through a 20-watt Power Amplifier into the 2-KMC High Gain Antenna or the 2-kmc Omni Antenna. It is excited by a crystal controlled oscillator or the voltage controlled oscillator in the Receiver.

(b) DSIF Power Amplifier - The DSIF Power Amplifier shall be a microswave power tube, either traveling wave tube (TWT), voltage tuneable magnetron (TM), or Amplitron, which has a minimum output of 20 watts. It shall be used to boost the signal from the DSIF transmitter to the 2-kmc Antenna.

3.3.1.2.1.4 Personal Communications.- Personal communications requirements within the confines of the C/M shall be met by helmet earphones and microphones during pressure suit operations and by separate headsets during shirtsleeve operations. Headset and bioinstrumentation outputs shall be transmitted via hardwire umbilicals from the crewmen to spacecraft systems.

Outside the spacecraft, a personal communications and telemetry system will be a part of the space suit assembly. Transmission from the space suit to the Spacecraft will be RF. Equipment within the space suit will be furnished by the space suit contractor and shall be electrically compatible with the NAA headset design. The command module VHF AM transceiver will be used for communications with the astronauts and will provide noninterfering two-way communications between the spacecraft and the crew members. The NASA furnished personal communications equipment shall include a headset, microphone and portable TV camera. Voice Communication shall be conducted at 243 mc with a 10 mw minimum output from the spacesuit assembly with optional use of a 5 watt maximum power amplifier. The personal communications system shall be capable of Television transmission at a frequency of 225-250 mc with a minimum output of 200 milliwatts.

3.3.1.2.1.5 C-Band Transponder.- The C-Band Transponder shall be used for radar tracking during earth-orbital missions and near-earth phase of lunar missions. It operates in the 5.4 to 5.9 kmc band through the C-band Antenna and shall be compatible with the AN-FPS-16 radar system. The C-band transponder accepts coded two pulse interrogations and provides single pulse replies. The transmitter shall have a minimum output of 2.5 kw peak power.

3.3.1.2.1.6 VHF Recovery Beacon. - The VHF Recovery Beacon may be used for direction finding during the recovery and post-landing phase. It operates at 243 mc into the VHF Recovery Antenna through the multiplexer and shall be compatible with direction finding systems presently available in rescue aircrafts. The beacon shall be automatically actuated after the extension of the VHF Recovery Antenna. The beacon shall be activated on command after earth landing.

3.3.1.2.1.7 HF Transceiver. - The HF transceiver may be used for long range voice communications and direction finding during the post-landing phase. It shall be all solid state and shall operate SSB, CW, or tone modulated in the 2 to 10 mc band. It shall be automatically actuated after earth landing and after deployment of the HF Recovery Antenna. The transmitter shall have a minimum output of 10 w PEP voice and 5 w peak tone.

3.3.1.2.2 Antenna Equipment Group. -

3.3.1.2.2.1 C-Band Antenna Equipment. - The C-Band Antenna shall be an array of flush mounted equispaced antennas and shall be used with the C-band Transponder. An amplitude comparator technique will be utilized to obtain an efficient antenna pattern.

3.3.1.2.2.2 VHF Broad Band Antenna. - The VHF Broad Band Antenna shall be used for all VHF signals that pass through the multiplexer. It shall be used during all near-earth and earth-orbital phases until the recovery phase. It shall be jettisoned prior to Drogue parachute deployment.

3.3.1.2.2.3 VHF Recovery Antenna. - The VHF Recovery Antenna shall be an extendable VHF antenna which is automatically extended after the main parachute is deployed. It shall be used for all VHF signals that pass through the multiplexer during the recovery and post-landing phases.

3.3.1.2.2.4 Back-up VHF Recovery Antenna. - The Back-up VHF Recovery Antenna shall be an extendable VHF Antenna deployed after the earth landing. It shall be used for all VHF signals that pass through the multiplexer in the event the VHF Recovery Antenna is not usable.

3.3.1.2.2.5 HF Recovery Antenna. - The HF Recovery Antenna shall be a sixteen foot HF whip which is deployed after earth landing. It shall be used with the HF recovery Transceiver during the post-landing phase.

3.3.1.2.2.6 2-kmc High Gain Antenna. - One High Gain Antenna shall be stored on opposite sides of the Service Module and shall be extended when in normal operation. Gimbal mounts permit rotation of the antenna for complete spherical coverage. The High Gain Antenna shall be used with the DSIF Transponder Equipment.

3.3.1.2.2.7 2-kmc Omni Antenna. - The 2-kmc Omni Antenna mounts on the VHF Broad band antenna (Discone) and shall be a slot array used for transmission of television data via the DSIF transponder equipment during earth orbital missions. It can be used during lunar missions in an emergency.

3.3.1.2.3 Intercommunications Equipment Group. -

3.3.1.2.3.1 Audio Center. - There shall be four Audio Centers located in the Electronic Installation Rack. Each center shall contain an earphone amplifier, a microphone amplifier, and VOX circuits. The earphone and microphone amplifier shall have AVC to minimize manual volume control. The crew member at any one station can select to talk on the intercom or on the communications link. The transmitter can be keyed either by VOX or by PTT.

3.3.1.2.3.2 Audio Control Unit. - There shall be four audio control units located in the Command Module and shall perform as described in paragraph 3.3.1.2.8.4.

3.3.1.2.3.3 Headsets. - Each headset shall contain dual microphones with a self contained transistor amplifier and dual earphones. One headset shall be provided for each crew member. A headset can be connected through the intercom to any voice receiver and/or voice transmitter without interfering with the other headsets.

3.3.1.2.4 Data Acquisition Equipment Group. -

3.3.1.2.4.1 Sensor Equipment. -Sensor equipment measurements shall include the following physical parameters:

(a) Temperature

(b) Pressure

(c) Flow

(d) Volume

(e) Leak Rate

(f) Acceleration

(g) Force

(h) Vibration

(i) Displacement

(j) Angular Velocity

(k) Gas Partial Pressure

(l) Acoustic Noise

(m) Stress/Strain

(n) Char and Ablation

3.3.1.2.4.2 Bio-medical Equipment (NASA Supplied). - The bio-medical equipment will measure various bodily functions of the astronauts. The required sensors and pre-amplifiers will be supplied by NASA. Examples of NASA supplied equipment includes:

(a) Sphygmomanometer

(b) Clinical Thermometer

(c) Gas Chromatograph

(d) Metabolic Balance Device

(e) Pulmonary Function Device

(f) Cardio-Vascular Measuring Device

(g) Stethoscope

3.3.1.2.4.3 Radiation Detection Equipment. - The radiation detection equipment shall monitor radiation rates and total dosages of various types of particles.

3.3.1.2.4.3.1 Personal Dosimeters (NASA Supplier). - Each crew member shall be supplied with a personal dosimeter system capable of measuring cumulative dosage. Each system shall contain a warning device and have an output plug for telemetry signals.

3.3.1.2.4.4 Scientific Instrumentation Equipment (NASA Supplied). - Various scientific instrumentation equipment will be installed in the Spacecraft.

3.3.1.2.4.5 Photographic Equipment. - The photographic equipment shall consist of one 16 mm motion picture camera for the control station panel display, and attachment to the telescope for real time motion events and one 70 mm pulse camera shall be included for documentation of Earth, Lunar and Terrestrial Bodies.

3.3.1.2.4.6 Optical Equipment. -The optical equipment consists of a telescope which shall be used for photographic recording, TV documentation and viewing (at unity and 20 power) of lunar and earth surfaces, landing sites, rendezvous, and docking. Two fields of view (3 and 60 degrees) are provided, with the larger magnification giving a resolution of 10 feet at 100 miles.

3.3.1.2.4.7 Television Equipment. -

(a) On Board Video Cameras - Two interchangeable video cameras with a repetition rate of 10 to 15 frames per second will provide real time video information for both the crew and earth-based receiving stations. Video information will be transmitted directly to earth at from 260 to 525 lines per frame via the DSIF transmitter for observation of astronauts and flight operations.

(b) Television Monitor -Listed under Communications and Instrumentation Controls and Displays Equipment Group, paragraph 3.3.1.2.8.

(c) Remote Television Equipment - Portable camera capability through the use of a battery and portable transmitter attachable to either camera shall be utilized for remote video pickup and relay to earth.

3.3.1.2.5 Data Handling Equipment Group. -

3.3.1.2.5.1 Signal Conditioning Equipment. - Combination attenuators, dc and ac amplifiers, frequency converters, phase demodulators, and impedance converters are used to condition all analog sensor and electrical subsystem signals to a standard level prior to encoding, recording or display.

3.3.1.2.5.2 Data Patch Panel Equipment. - A patch panel shall be provided between signal conditioners, sensors and the telemetry equipment for the following purposes: (1) Provide flexibility in measurement programming. (2) Allow time sharing of both signal conditioners and analog commutators during a mission.

A patch panel shall be provided ahead of the digital commutator to permit the flexible selection of digital data.

3.3.1.2.5.3 Telemetry Equipment. - The Telemetry Equipment processes Spacecraft data into forms suitable for transmission to earth by means of a radio link. Conditioned analog and sensor data are commutated and digitized into 8 bit words. These words, on-off functions and computer digital data format are transmitted to earth during deep space phases of lunar missions via the DSIF transponder. During near-earth phases of all missions, data shall be transmitted over the VHF Transmitter. Accuracy of the PCM data shall be 0.5 percent not including the transducer. Minimum input impedance shall be 500K ohm. Bit rate shall be fixed at 64,000 bits/sec with an NRZ format. Telemetry signal inputs will vary between 0 and 5 volts or 0 and 250 m volts with 5000 ohm (max) source impedances. Functions of the Telemetry Equipment are:

(a) Analog Commutation

(b) A-D Conversion

(c) Digital Commutation

(d) Programming

(e) Calibration

3.3.1.2.6 Data Storage Equipment Group. - A magnetic tape recorder will be provided for on-board storage and reproduction of analog and digital data. PCM, voice or analog data can be recorded during critical periods and/or communication blackouts and played back later for transmission to earth or can be stored for playback after recovery. Fourteen channels on the recorder are available:

1 Channel - Spacecraft timing pulse used for decoding digital data upon playback

4 Channels - Each capable of recording 32,000 bits/sec of NRZ digital data

9 Channels - Analog -50 cps to 10,000 cps

3.3.1.2.6.1 Tape Deck. - The tape transport unit provides for 30 minutes record time, forward and reverse modes, and fast rewind.

3.3.1.2.6.2 Recorder Electronics. - The recorder electronics provide the capability of recording, erasing or playback of the magnetic tape. The recorded clock track shall be used during playback to provide a compatible time-base for the digital data.

3.3.1.2.7 Spacecraft Central Timing Equipment Group. - The Spacecraft Central Timing Equipment shall be used for timing the various operations in the Spacecraft. The reference frequency standard oscillator shall be accurate to ±0.5 second in 14 days. Basic operating frequency is 512K cps. A series of dividers are used to obtain the various frequencies. Synchronization shall be obtained from guidance and navigation subsystem. The Central timing equipment provides synchronization, to the Television equipment, to the Cameras, to the PCM Telemetry equipment, to the Calibration equipment, to the electrical power to Recorders, Television equipment, Cameras, PCM Telemetry equipment, Calibration equipment, the electrical power subsystem and the displays on the Display Panel. The elapsed time shall be derived from the count-down frequencies for crew display. This time shall be also pulse width coded and conditioned for recording on the tape recorders, television equipment, and cameras. The PCM telemetry equipment also derives its reference frequency from the timing equipment.

3.3.1.2.8 Communications and Instrumentation Controls and Display Equipment Group. -

3.3.1.2.8.1 Main Control Panel Components. - The main Control Panel shall provide the following control capabilities.

(a) C-Band Transponder Tracking Switch

(b) Rescue HF Mode Selector

(c) Rescue VHF Beacon Switch

(f) TM Data Selector

(h) Data Recorder Mode Selector

(i) Tape Operation Selector

(j) Tape Direction Selector

(k) Personal Communications Selector

(l) TV Earth Relay Selector

In addition to these control capabilities, the main control panel shall provide the following display:

Recorder Indicator: Five Minute Warning and End of Tape

3.3.1.2.8.2 DSIF/Near-Earth Control Panel Components. - The DSIF/Near-Earth Control Panel shall provide the following control capabilities:

(a) DSIF Power Amplifier Output Power Selector

(b) DSIF Modulator Selector

(c) DSIF Mode Selector

(d) Near-Earth VHF FM Transmitter Switch

(e) VHF AM Transceiver on-off Switch

(f) DSIF OSC Selector

3.3.1.2.8.3 Antenna Control Panel. - The Antenna Control Panel shall provide the following control capabilites.

(a) Antenna Position Meters for: (1) Yaw
(2) Pitch

(b) AGC Level Meter

(c) Antenna Position Pitch Switch

(d) Antenna Position Yaw Switch

(e) Drive Mode Selector

(f) Antenna Deployment Switches (2 required)

(g) Antenna Selector

3.3.1.2.8.4 Audio Control Unit. - The audio control units (4 required) shall be located at each control station and the work station and provide switching to enable the crew to select any Spacecraft communication link on an interference-free basis. All of the circuits are selectable in either transmit or receive, or receive only modes - VOX or PTT operation shall be selectable. Each Audio Control Unit shall contain its own volume control.

3.3.1.2.8.5 Television Monitor Display. - The television picture monitor display shall be located on the display panel. It shall be used for crew monitoring of the output of either camera including scenes being transmitted.

3.3.1.2.8.6 Spacecraft Central Timing Equipment Display. - The digital type displays present Time in GMT, Time to Event, Time After Event, and Total Elapsed Time.

3.3.1.3 Component Location. - The general area locations for the major elements of the Communications and Instrumentation System are indicated on Figure 3.

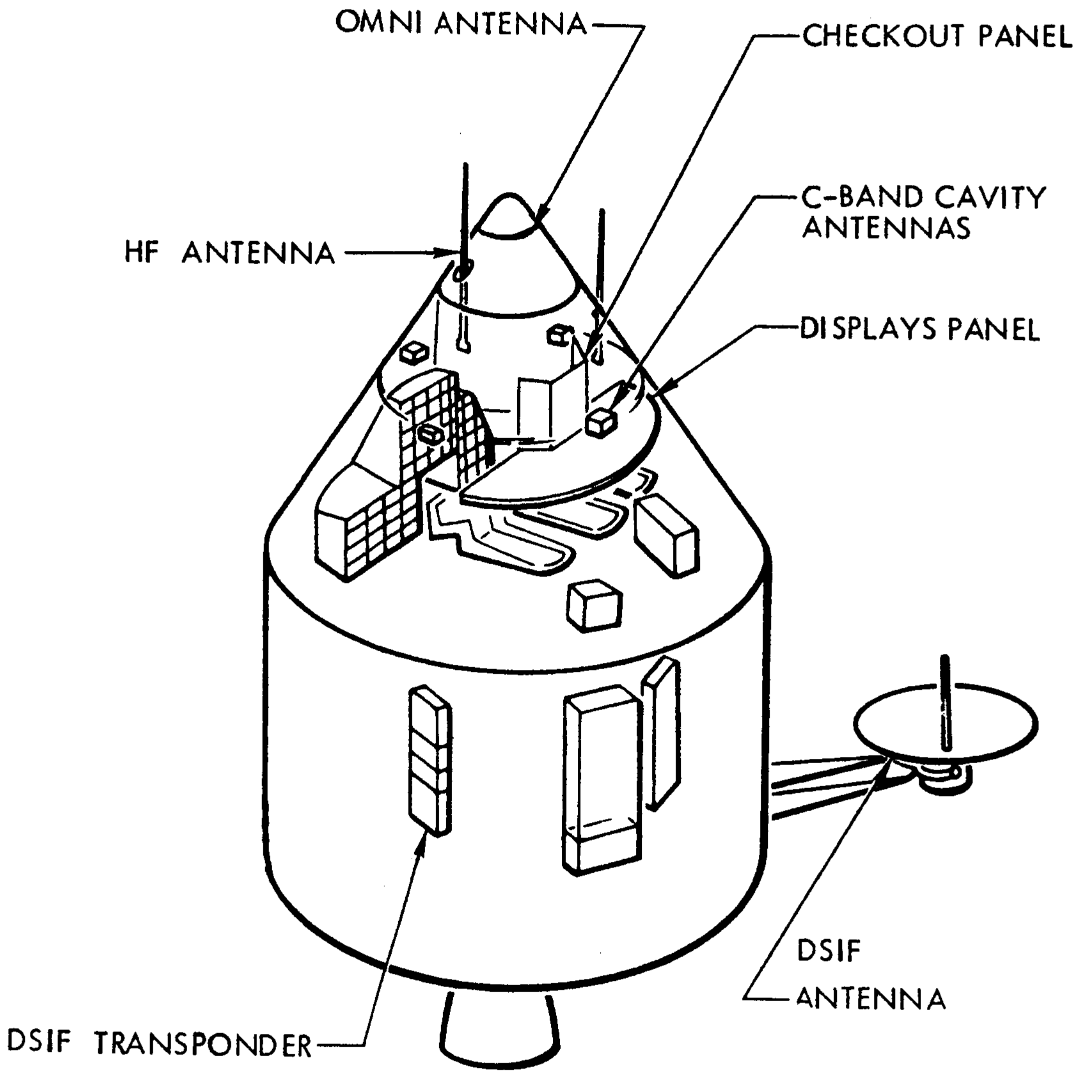

Figure 3. Apollo Communications System

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3.3.2 Electrical Power System (EPS). -

3.3.2.1 System Performance. - The primary power source for the Apollo spacecraft shall be a fuel cell system (Reference figure 4 Electrical Power System Block Diagram). The fuel cell system shall consist of three fuel cell modules and mechanical accessories.

3.3.2.2 System Components. -

3.3.2.2.1 Fuel Cells. - Three fuel cell modules shall provide the main power throughout the mission until S/M separation. Each fuel cell battery shall consist of a number of non-regenerative hydrogen-oxygen fuel cells connected in series and capable of producing 1500 watts. The fuel cells are of the low-pressure, intermediate-temperature, Bacon type, using potassium hydroxide electrolyte.

3.3.2.2.2 Reactant Tanks. - Reactants are stored in the supercritical cryogenic state in four tanks, two for each reactant as shown in Figure 5. This system also furnishes oxygen to the ECS.

3.3.2.2.3 Space Radiator. - A space radiator system with two radiator panels shall be designed to dissipate waste heat generated in the fuel cells. Condensed water will be separated and supplied to the environmental control system in potable form.

3.3.2.2.4 Storage Batteries. - Auxiliary electrical power shall be provided by three silver zinc storage batteries. Two of these shall supply all power during entry and recovery flight phases, as well as supplementary power during peak load periods on the fuel cell system. The third battery shall be isolated from the electrical power system and supply only post-landing loads. A battery charger shall be provided to recharge any of the three batteries after normal or emergency discharge.

3.3.2.2.5 Circuit Protectors. - Circuit protection for the power distribution system shall be provided by a system of circuit breakers and diodes so that power will not be interrupted to essential loads due to short circuits in any power source, any feeder circuit, either essential bus, or any load.

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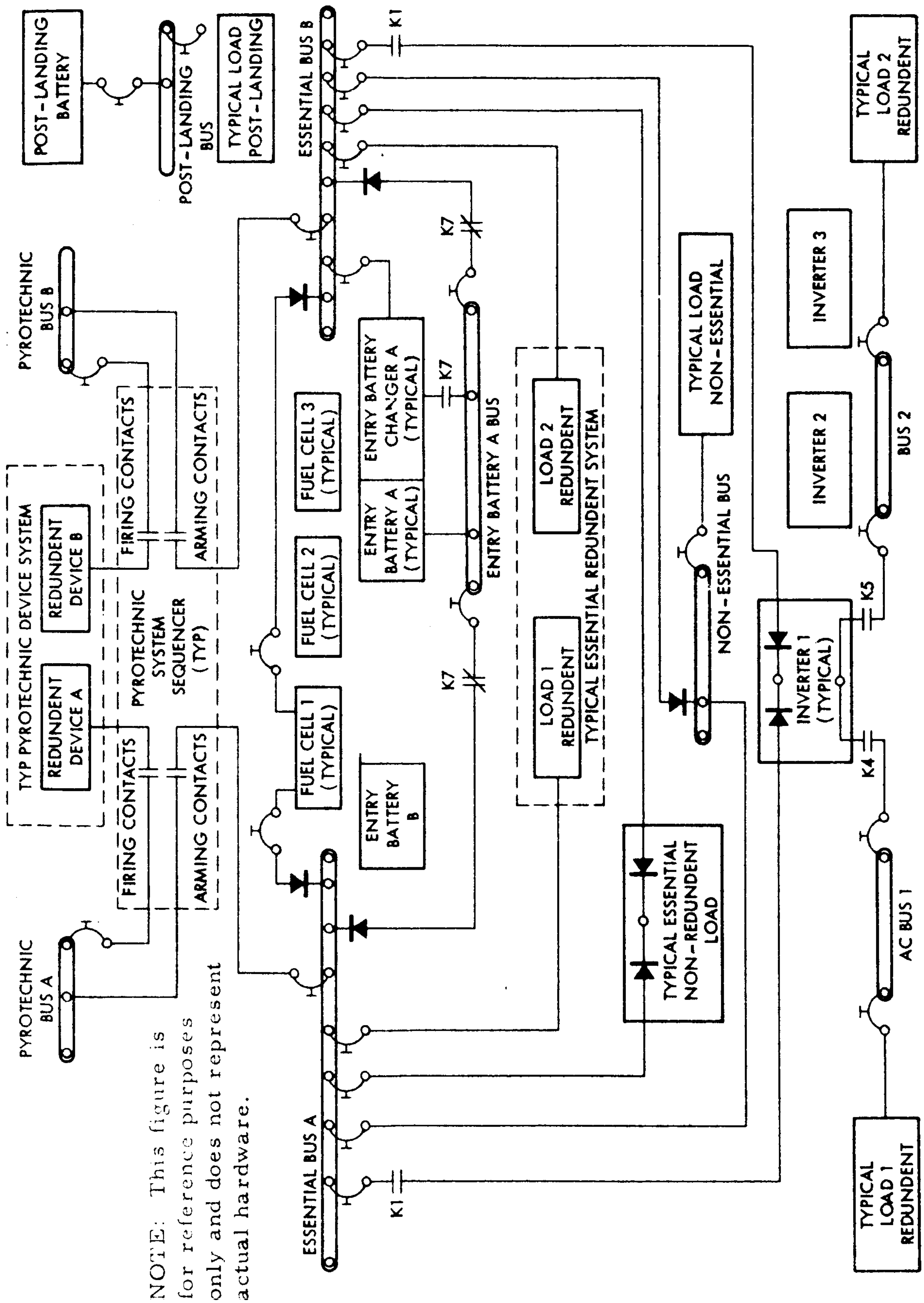

NOTE: This figure is for reference purposes only and does not represent actual hardware.

Figure 4. Electrical System Block Diagram

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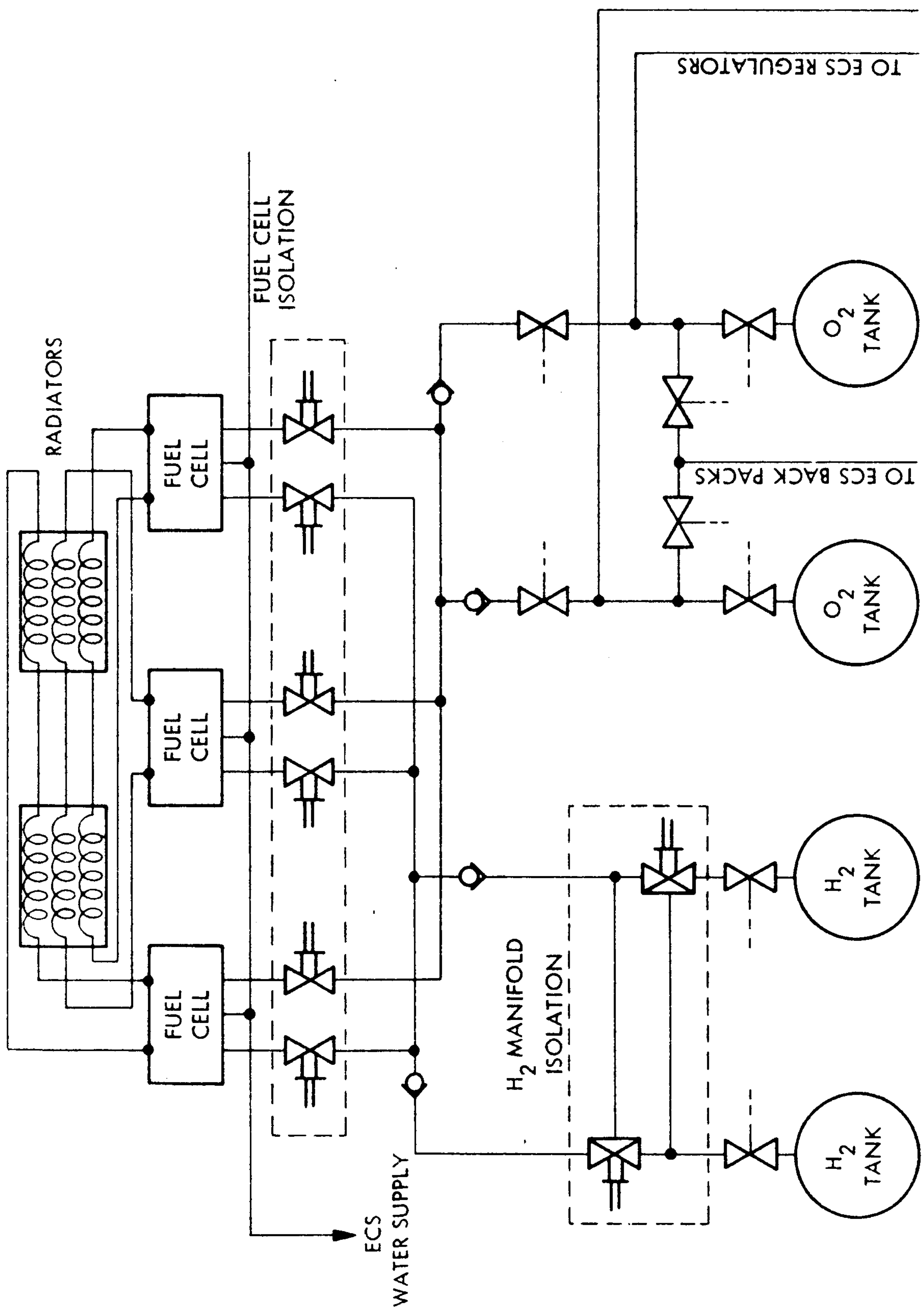

THIS FIGURE IS FOR REFERENCE ONLY AND DOES NOT REPRESENT ACTUAL HARDWARE

Figure 5. Cryogenic Supercritical Gas Supply To Fuel Cells

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3.3.2.2.6 Inverters. - Three static inverters shall supply 400 cps, three-phase, Y connected, ac power. One inverter only will operate with the other two acting as idle standby units.

3.3.2.2.7 Mechanical Accessories. - Mechanical accessories for the system shall include heat exchangers, water extractors, circulators, valves and piping.

3.3.2.3 Distribution. -

3.3.2.3.1 Pre-Launch Power Distribution. - GSE dc electrical power shall be received by the service module electrical power system to replace the use of fuel cells and batteries during preflight system checkout. Power demands on the ground power source vary anywhere from zero to 4,000 watts.

Electrical power characteristics of ground power at the spacecraft load shall be the same as provided by the fuel cells. To provide voltage within these limits at the load, ground power should be maintained within the limits of 27 to 31 volts at the umbilical for steady-state and transient conditions.

External dc power shall be provided to the command module through two separate feeders to Essential Buses A and B. Each feeder shall be capable of alternately being energized to check the spacecraft Essential Bus isolation diode system. However, both feeders operating in parallel will be used during maximum load conditions.

The external dc power shall not ground the spacecraft negative bus.

No external ac power shall be required for preflight system checkout.

External dc power will be used for this purpose, supplying ac through the spacecraft inverter.

3.3.2.3.2 Distribution Within Spacecraft. - All fuel cell modules and both entry batteries shall be connected to a dual, redundant, essential bus system supplying all essential loads redundantly, so that power to all essential loads systems shall be maintained in the event of failure to one or two fuel cell modules, one entry battery, and/or one essential bus. Non-essential loads shall be connected to a separate bus, which may be disconnected from the essential bus in the event of failure of two fuel cells or one entry battery. Essential loads are defined as those essential to safety of flight, and non-essential loads as those necessary to successful completion of mission.

The command module electrical power system shall receive dc power from the service module power supply and shall then control and distribute this power to all spacecraft systems requiring dc electrical power. The command module electrical power system shall convert some dc power to ac power and shall then regulate, control, and distribute this ac power.

3.3.2.4 Operation Modes. -

3.3.2.4.1 Normal Mode. -

(1) Three fuel cell modules operative: These cells supply power for all normal loads until S/M separation.

(2) One Fuel Cell Module inoperative: The remaining two cells supply power for all normal loads until S/M separation.

3.3.2.4.2 Emergency Mode. -

(1) Two fuel cell modules inoperative: All non-essential loads shall be removed. The remaining single fuel cell module supplemented by the entry batteries, supplies power to all essential loads.

3.3.2.5 System Loads. -

3.3.2.5.1 Essential Loads. - Electrical loads required to insure the safe return of the crew under the emergency operating mode of the electrical power system are defined as essential loads.

3.3.2.5.2 <u>Non-Essential Loads</u>.- Electrical loads not required to insure the safe return of the crew under the emergency operating mode of the electrical power system.

3.3.2.6 <u>Electrical Voltage Characteristics</u>.-

3.3.2.6.1 <u>DC Voltage Characteristics</u>.-

(1) Steady State Load Voltage Limits: 25-30 volts.

(2) Steady State Fuel Cell Output Voltage Limits: 27-31 volts.

(3) Transient Load Voltage Limits: 25-32 volts, 0.7 sec. recovery.

(4) Emergency Load Limit: 20 volts (min).

(5) Load Voltage Ripple: 250 millivolts (max) peak to peak.

(6) Maximum Fuel Cell Output: 1500 watts (each) at 27 volts.

3.3.2.6.2 <u>AC Voltage Characteristics</u>.-

(1) Phases: 3 phase displaced 120 ±2 degrees, Y connected, with phase rotation of A-B-C referred to the bus phase designation.

(2) Steady State Voltage Limits: 115 ±2 volts rms, average of 3 phases.

(3) Transient Voltage Limits: 105-125 volts rms, 0.05 sec. recovery.

(4) Voltage Unbalance: 2 percent max., deviation of worst phase from average.

(5) Frequency: 400 cps, synchronized to spacecraft clock. For emergency operation, in the event of loss of spacecraft sychronizing signal, the steady state and transient frequency limits shall be 390 ± 10 cps.

NOTE: Clock accuracy from primary source (Guidance and Navigation System) is one part in ten million for period of 14 days. Accuracy from secondary backup source (Communication and Instrumentation System) is two parts on one million for period of 5 days. Probability of success of the combined primary and secondary system is 0.9999.

(6) Frequency Modulation: Steady State ±0.25%.

(7) Maximum output of each inverter: 1 KVA.

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3.3.2.7 Fuel Cell Reactant Characteristics. -

(1) Total Reactant Weight: 80 H_2
580 O_2

(2) Fuel Consumption: 0.90 lb/kw-hr average total O_2 and H_2 (under normal conditions)

3.3.2.8 Component Location. -

Component	Location
Fuel Cells	S/M
Space Radiators	S/M
Heat Exchangers, Piping, Valves	S/M
Reactant Tanks	S/M
Zinc-silver Oxide Batteries	C/M
Electrical Power Distribution and Controls	C/M
Inverters	C/M

3.3.3 Environmental Control System (ECS). -

3.3.3.1 Function. - The function of the ECS (reference Figure 6) shall be to provide a control of the flow, pressure, temperature, and composition of the gases for the shirtsleeve atmosphere of the Command Module. These gases shall also be utilized in the operation of the three pressure suits used by the crewmen during the escape and entry phases, emergency conditions, and with the backpacks during lunar exploration. The ECS system shall provide thermal control of the Command Module and the equipment used in the Command and Service Modules where required. System operation shall be automatic with a provision for manual control by the flight crew in the event of emergency. Facilities shall be provided for charging the backpacks (NASA-furnished)

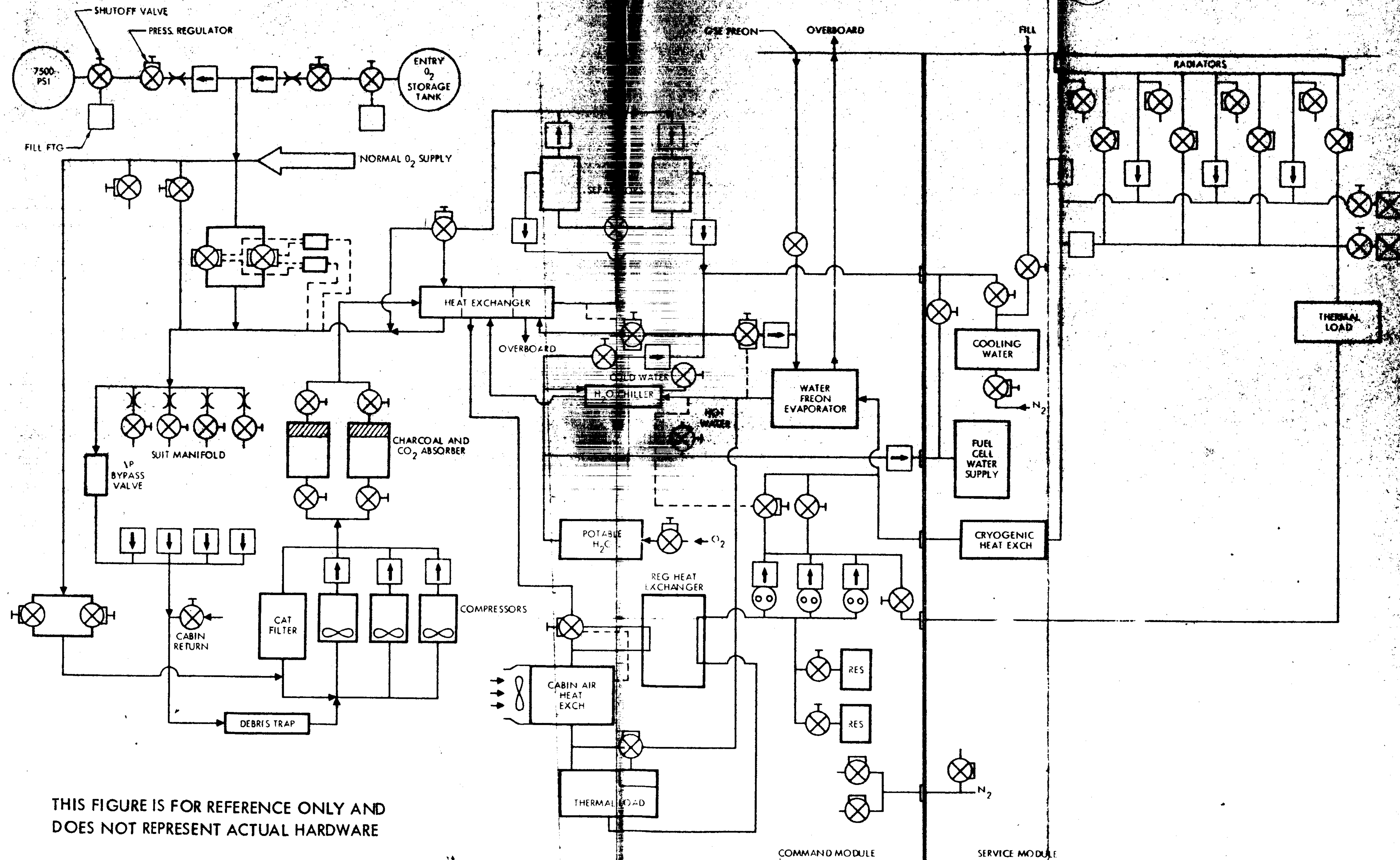

Figure 6. Environmental Control System (Tentative)

which are self-contained, extravehicular environmental control systems to be used in conjunction with the pressure suits for lunar exploration and emergency conditions. A water management system shall be provided to be used for crew water consumption and emergency heat control operations. The environmental control system may provide some additional individual nuclear radiation protection over and above that afforded by the Spacecraft structure as necessary.

3.3.3.1.1 Pre-Launch Requirements.- The ground support equipment shall be compatible with the cooling equipment in the spacecraft environmental control system and shall assume the requirement of cooling the internal heat load of the Spacecraft during the pre-launch phase of the mission.

3.3.3.2 Performance.- The performance of the ECS shall include removal of the carbon dioxide and various odors by lithium hydroxide and activated charcoal or other suitable means. Atmospheric make-up fluids shall be obtained from supercritically stored oxygen and nitrogen. Trace contaminant control shall be accomplished by a catalytic filter. Equipment and crew heat loads shall be rejected to space through a radiator, and emergency cooling shall be provided by evaporating water. The ECS shall provide for water management. The primary source of water shall be the water produced as a byproduct of the electrical power generation system. The ECS shall also provide a supply of water for controlling the total internal heat load from the entry interface to 100,000 ft altitude. From 100,000 feet to main chute deployment the ECS shall provide air circulation only. From main chute deployment to touchdown and during the post landing phase, the ECS shall provide for cabin ventilation with outside air only.

The ECS shall be capable of maintaining the following Command Module performance characteristics:

Function	Normal	Emergency
(a) Cabin pressure (psia)	5.0 to 15.0	3.5 (min) for 5 minutes with a 0.5 inch hole
(b) Cabin temperature (°F)	75 ± 5	40 - 75
(c) O_2 Partial Pressure (mm Hg)	233 (nominal)	0*
(d) CO_2 Partial Pressure (mm Hg)	7.6 (max)	0*

*Compartment decompression

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Function	Normal	Emergency
(e) Relative Humidity (%)	40 to 70	0*
(f) O_2 output rate (lb/day)	5.4	50% reserve
(g) CO_2 input absorption rate (lb/day)	6.9	15% overload capability
(h) Water production (lb/hr)	1.1 to 2.3 (varies with electric load)	0.56 to 1.0 (electrical emergency)
(i) Supply Pressure (psia)	7 - 63	7 - 63
(j) Total heat absorption rate (BTU/hr)	10,000 (max)	

3.3.3.3 Subsystem.- The ECS shall consist of six subsystems as described in the following paragraphs:

3.3.3.3.1 Pressure Suit Circuit Subsystem.- The pressure suit circuit shall automatically control the flow, pressure, temperature, and composition of the pressure suit gas. This sytem, in conjunction with the Command Module pressure and temperature control subsystem, shall also control these environmental conditions in the Command Module when any or all of the crew are out of their pressure suits.

These functions shall be provided by the water separators, a regenerative heat exchanger, oxygen partial pressure controls, oxygen demand regulators, a debris trap, air compressors, carbon dioxide and odor absorbers, a catalytic filter, and air cooler, and a standby water evaporator. Component redundancies shall be provided as required to obtain the reliability suitable for the Apollo missions.

3.3.3.3.2 Water-Glycol Circuit Subsystem.- The water-glycol circuit subsystem shall be a closed, intermediate, heat-transfer loop which permits heat to be delivered from the space vehicle interior to the two space radiator panels. This subsystem shall provide cooling for electronic and electrical equipment. This function shall be accomplished by pumps, heat exchangers, valves, and controls.

3.3.3.3.3 Command Module Pressure and Temperature Control Subsystem.- This subsystem shall automatically maintain the pressure and temperature of

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the Command Module interior within prescribed limits. This function shall be accomplished in conjunction with the pressure suit circuit subsystem by means of regulated inflow, recirculation blowers, a heat exchanger, a temperature control and sensor snorkel valves, Command Module pressure relief valve, and various other valves and controls. This subsystem shall permit charging of the backpacks within the Command Module.

3.3.3.3.4 Oxygen Supply Subsystem.- The oxygen supply subsystem shall be subdivided into three systems:

(a) the normal oxygen supply system,

(b) the partial pressure control system,

(c) the entry oxygen supply system.

3.3.3.3.4.1 Normal Oxygen Supply System.- The normal oxygen supply system shall be capable of supplying all of the required oxygen for the entire mission up to the entry phase. The system shall function by reducing the pressure of the space vehicle oxygen supply system as required to the nominal working pressure of the various suit circuit and potable water tank pressure controls. This function shall be accomplished by suitable pressure regulators, demand regulators, and shutoff and check valves.

3.3.3.3.4.2 Partial Pressure Control System.- The partial pressure control system shall monitor the oxygen partial pressure within the pressure suit loop and Command Module interior and maintain the oxygen partial pressure at a predetermined minimum level. This function shall be accomplished by partial pressure sensors, partial pressure controls, demand pressure regulators for 3.5 psia operation, and inflow control valves.

3.3.3.3.4.3 Entry Oxygen Supply System.- The entry oxygen supply system shall supply the oxygen required for mission completion after separation of the Service Module from the Command Module. This system shall store gaseous oxygen at a high pressure. During system operation, the high pressure shall be reduced to the nominal working pressure of the various suit circuit pressure controls. This function shall be accomplished by oxygen storage tanks, high and low pressure regulators, and valves.

3.3.3.3.5 Water Supply Subsystem.- In addition to furnishing potable water for consumption by the crew members, the water supply subsystem shall furnish supplementary cooling of the water-glycol heat transfer loop in the event of insufficient heat dissipation by the spare radiators and emergency cooling of the suit loop. Potable water shall be stored prior to take-off and shall be replenished from the fuel cell. These functions shall be accomplished utilizing water tanks, pressure controls, and valves. This subsystem shall contain several valves associated with other subsystems: i.e., Freon launch cooling, glycol circuit, suit circuit, and oxygen supply circuit.

3.3.3.4 Component Location.- Refer to figure 6.

3.3.4 In-Flight Test System (IFTS).- An IFTS may be incorporated to improve the overall mission reliability figure.

3.3.4.1 Function.- This system shall provide the crew with a rapid check on the status of the Spacecraft systems and also provide a means of isolating the malfunction to the module level.

3.3.4.2 Performance.- This concept of the In-Flight Test System provides a GO-NO GO test point readout, a coded readout showing which system and subsystem failed, a means for manual testing, maintenance instructions, and a centralized panel which can be used for ground checkout and telemeter data needs. This concept is illustrated by means of a functional block diagram shown in Figure 7.

3.3.4.3 IFTS Subsystems.- The principal subsystems are as follows:

(a) Central Panel

(b) Programmer

(c) Switching Complex

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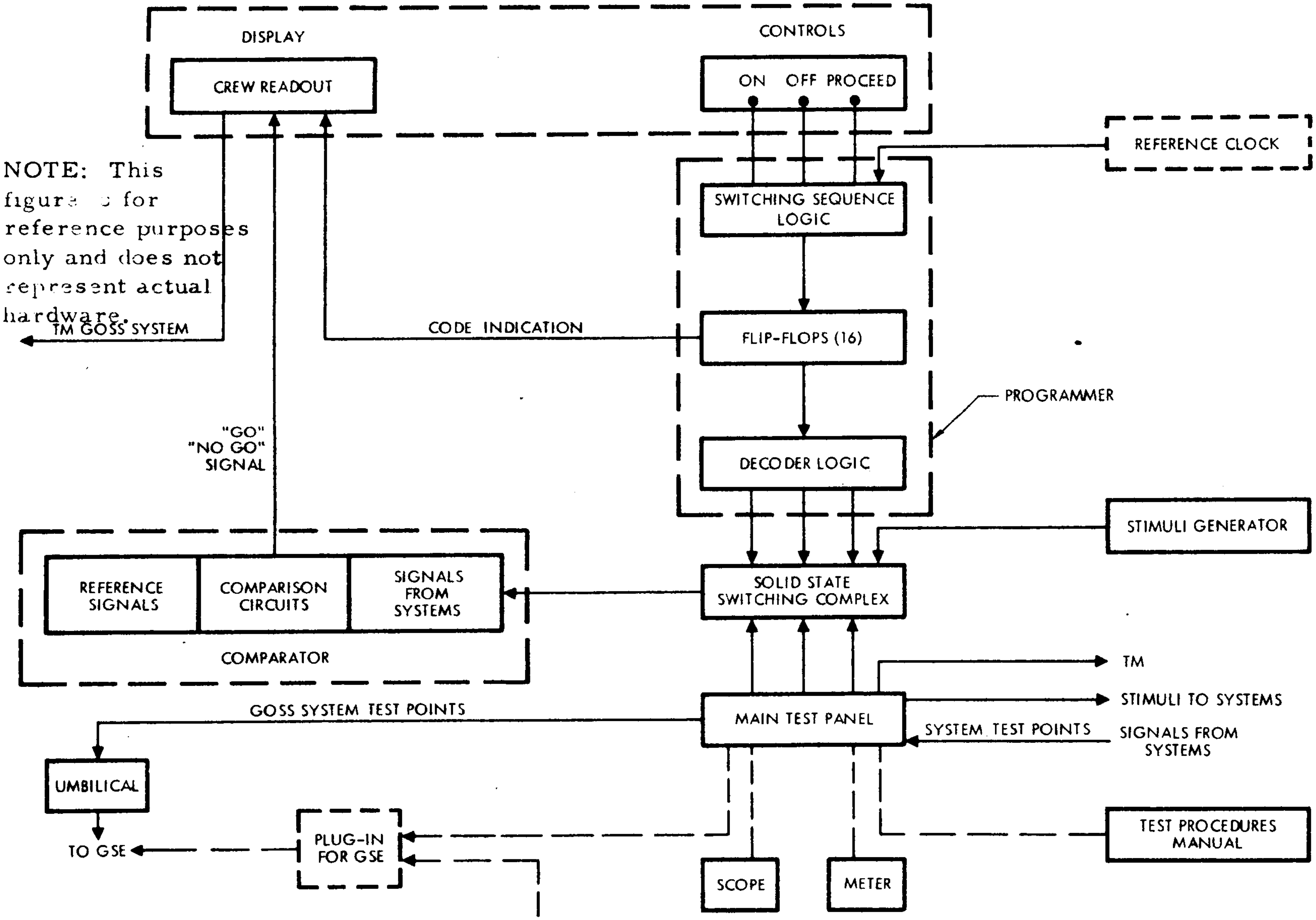

NOTE: This figure is for reference purposes only and does not represent actual hardware.

Figure 7. Block Diagram of a Typical In-Flight Test System

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(d) Comparator

(e) Crew Readout

(f) Manual Test Unit

(g) Stimuli Generator

3.3.4.3.1 Central Panel.- The central panel is the termination area for all test point wiring originating in the Spacecraft systems which will be monitored by the IFTS or used in Manual Testing.

3.3.4.3.2 Programmer.- The programmer is a sequential device which controls the Switching Complex and Comparator.

3.3.4.3.3 Switching Complex.- The Switching Complex connects the test points to the Comparator. The proper standard comparator voltages for the test points being monitored are also switched to the Comparator.

3.3.4.3.4 Comparator.- The Comparator then senses the test signals and standard voltages, compares them, and the result of the comparison is read out as a GO-NO GO on the Crew Readout.

3.3.4.3.5 Crew Readout.- The Crew Readout provides a GO-NO GO indication of the system or subsystem condition and a coded manual test reference, for use in performing manual tests with the Manual Test Unit, or for finding maintenance instructions.

3.3.4.3.6 Manual Test Unit.- The manual test unit is comprised of a scope and meter, and is used in making limited diagnostic tests through test points accessible at the Central Panel. This is the first step in the maintenance of the Spacecraft systems below the GO-NO GO indication of the Crew Readout.

3.3.4.3.7 Stimuli Generator.- The Stimuli Generator is used for two purposes:

(a) For supplying stimuli and power to the non-operating redundant circuits during the GO-NO GO sequential tests.

(b) For supplying stimuli to the subsystems or module under test to achieve a test point output.

3.3.5 Structural System. -

3.3.5.1 General. -

3.3.5.1.1 Design Factors. -

3.3.5.1.1.1 Ultimate Factor. - The ultimate factor shall be 1.5 applied to limit loads. This factor may be reduced to 1.35 for special cases upon rational analysis and negotiation with the Manned Spacecraft Center.

3.3.5.1.1.2 Pressure Vessel Design Factors. - Pressure vessels shall be designed using the following factors based on limit loads.

3.3.5.1.1.2.1 Pressure Vessel Proof Factor. - The proof factor shall be 1.33 when the pressure is applied as a singular load. This factor may be reduced for special cases upon rational analysis and negotiations with the Manned Spacecraft Center.

3.3.5.1.1.2.2 Pressure Vessel Ultimate Factors. - The ultimate factor shall be 1.50 when pressure is applied as a singular load.

3.3.5.1.1.2.3 Pressure Vessel Limit Loads. - Limit loads shall be obtained with limit pressures.

3.3.5.1.1.2.4 Pressure Stabilized Structures. - No structure shall require pressure stabilization.

3.3.5.1.1.3 Hydraulic or Pneumatic Systems. - Flexible hose, tubing and fittings less than 1.5 inches in diameter shall have a proof pressure of 2.00 times the limit pressure and a burst pressure of 4.00 times the limit pressure. Flexible hose, tubing, and fittings greater than 1.5 inches in diameter shall have a proof pressure of 1.50 times the limit pressure and a burst pressure of 2.50 times the limit pressure.

3.3.5.1.1.4 Air Reservoirs. - Air reservoirs shall have a proof pressure of 1.5 times the limit pressure and an ultimate pressure of 2.5 times the limit pressure.

3.3.5.1.2 Flight Loads. -

3.3.5.1.2.1 Tumbling at Maximum Dynamic Pressure. - The Command Module-Launch Escape System combination shall be designed for loads arising from tumbling of the escape vehicle at maximum dynamic pressure during boost.

3.3.5.1.2.2 Entry. - The Command Module shall be designed for a limit load of 20 g during entry.

3.3.5.1.2.3 Noise. - The design shall accommodate exterior sound pressure levels of 171 db in the frequency range of 4 to 9600 cps emanating from the Launch Escape System during both launch and abort modes. The noise level within the Command Module shall be within limits given in Figure 8.

3.3.5.1.2.4 Buffet. - The design shall accommodate a limit buffet pressure of 1.5 psi (RMS) in the frequency range of 0 to 4 cps on the spacecraft and Adapter during the earth launch phases.

3.3.5.1.2.5 End Boost Stage 1. - Calculation of transverse loads on the space vehicle shall consider two engines deflected hard over against the stops. The transient phenomena associated with thrust decay shall also be investigated.

3.3.5.1.2.6 Stage Separation. - The separation of the lower stages shall be accomplished with no subsequent re-contact and resulting dynamic load input to the spacecraft.

3.3.5.1.2.7 Boost of Upper Stages. - Structural loads during Launch Vehicle upper stage operation shall include maximum control deflection of thrusting engines balanced by rotary inertiaand steady-state and dynamic load associated with engine-out operation. Space vehicle structural dynamic response to thrust transients at thrust build-up and tail-off shall be investigated, including effects of vehicle flexibility and its interaction with liquid propellants.

3.3.5.1.2.7.1 Jettison of Launch Escape System. - The release actuation of the Launch Escape System shall not be impaired during imposed load factors of 2.0 g longitudinal or 1.0 transverse, applied individually or in combination to the Spacecraft. All disconnect mechanisms shall avoid release of parts that may pierce tanks or inflict damage to vital components.

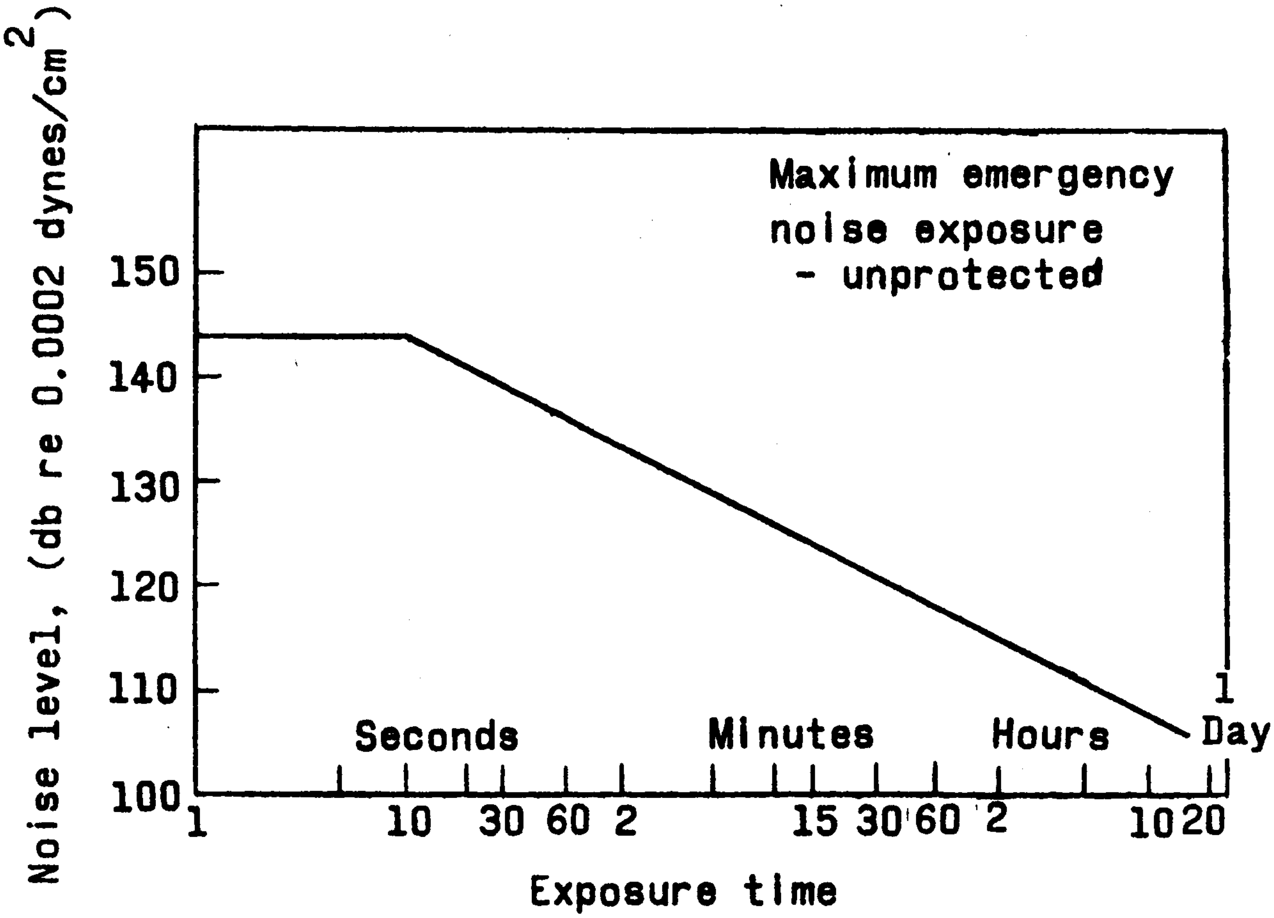

Figure 8. Noise Tolerance, Emergency Limit.

3.3.5.1.2.8 Vibration. - The application of propulsion system transients shall consider the engines to be deflected in the worst manner within allowable gimbal limits. The effects of the steady and transient inputs shall be combined. The vibration analyses shall recognize the lower damping present in a vacuum.

3.3.5.1.2.9 Dynamic Loading. - The calculation of dynamic loads shall include the effects of engine start, rebound on the pad, lift off transients including ground winds, gusts, wind shears, buffeting, and longitudinal resonance. The coupling of the structural dynamics with the flight control system shall be included in the determination of dynamic loads.

3.3.5.1.2.10 Separation, Maneuvering, and Docking. - Separation, maneuvering and docking loads associated with the Spacecraft in orbit shall be considered.

3.3.5.1.2.10.1 Separation. - Separation shall not be impaired by reasonable tolerances on symmetry and/or simultaneity of thrust pulses of reaction nozzles. All disconnect mechanisms shall avoid release of parts which may pierce or inflict damage to vital components.

3.3.5.1.2.10.2 Maneuvering. - The loads shall reflect the response of the structure to the transient thrust forces.

3.3.5.1.2.10.3 Docking. - The loads and energy levels shall be defined by a dynamic analysis of the docking impact. This analysis shall also furnish load factors on crew, equipment, and structure.

3.3.5.1.2.11 Entry. - To conservatively account for the effects of venting lags in cabin pressure, internal cabin pressure shall be considered to be zero psia where critical during entry. Effects of temperature and temperature gradients due to aerodynamic heating on material properties, thermal stresses, and deformation shall be considered simultaneously with applied external loads and internal pressures.

3.3.5.1.2.12 Earth Landing. - The terminal impact attenuation system shall be compatible with the use of parachutes. After normal entry, maximum q escape, and pad escape, the landing system shall stabilize the Command Module during postentry descent and shall reduce the vertical velocity to not more than 30 feet per second at 5000 feet altitude. The landing system shall be designed to safely land the Command Module on land or water. Any further attenuation required by the crew shall be provided by energy-absorbing devices in the individual crew support and restraint systems. After landing, the landing system shall provide any necessary flotation, survival and location aids.

3.3.5.1.2.12.1 Parachute System. - The following loading conditions during earth landing shall be considered in structural design of the Command Module.

3.3.5.1.2.12.1.1 Actuation of Parachute. - Effects of geometry, direction of force application, inertia, and structural flexibility on vehicle loads shall be included.

3.3.5.1.2.12.1.2 Snatch and Opening Shock Loads. - Force transients from the drogues, pilot, and main parachutes due to snatch and opening shock shall be included.

3.3.5.1.2.12.1.3 Off-Axis Loading. - Dynamic analyses of module motion shall be performed to establish maximum design loads on individual parachute riser attachments and module accelerations. The influence of structural flexibility, where important, shall be included.

3.3.5.1.2.12.1.4 Internal Pressures. - The pressure cabin shall be designed to a maximum bursting pressure of 18 psia ultimate to cover the case where all cabin vents fail during boost. During entry up until the drogue parachutes are released, the internal cabin pressure shall be assumed to be zero psia. During landing impact, the minimum internal cabin pressure shall be assumed to be zero psig to cover the case where all cabin vents fail.

3.3.5.1.2.12.1.5 Entry Heating. - Entry heating effects upon structural properties, thermal stresses, and expansions shall be accounted for.

3.3.5.1.2.12.1.6 Normal Landing Impact. - Design requirements for normal landing impact shall be as follows:

(a) The Command Module shall not turn over during or after ground contact.

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(b) Nominal crew tolerances to impact accelerations and acceleration onset rates shown in Figures 9 and 10 shall not be exceeded during the dynamics of ground contact.

(c) The Command Module primary structure shall remain unimpaired following the application of dynamic loads during contact.

(d) Accelerations of the Command Module primary structure including flexibility effects shall not exceed 40 g.

To conservatively cover presently unknown factors related to landing site terrain, the following assumptions of terrain characteristics shall be made:

(a) Obstacles such as ravines, hummocks, rocks, boulders, bushes, and trees shall not be considered; however, the maximum slope in any direction shall be 5 degrees, and the surface shall be smooth and plane.

(b) The coefficient of surface friction shall be between 0.3 and 0.5.

(c) The maximum terrain altitude shall be 5000 feet above sea level.

A maximum horizontal steady wind velocity of 30 knots shall be assumed at 10 feet above ground level to conservatively cover the possibility of prevailing winds.

3.3.5.1.2.12.1.7 Roll. - For initial design purposes, roll angle shall be less than 5 degrees in either direction.

3.3.5.1.2.12.1.8 Impact Attitude. - The nominal impact attitude shall allow for a maximum pendulum-type oscillation of ± 8 degrees in any direction for two parachute operation. The tangential velocities incurred by this oscillation shall be included in the calculation of Command Module velocity at impact.

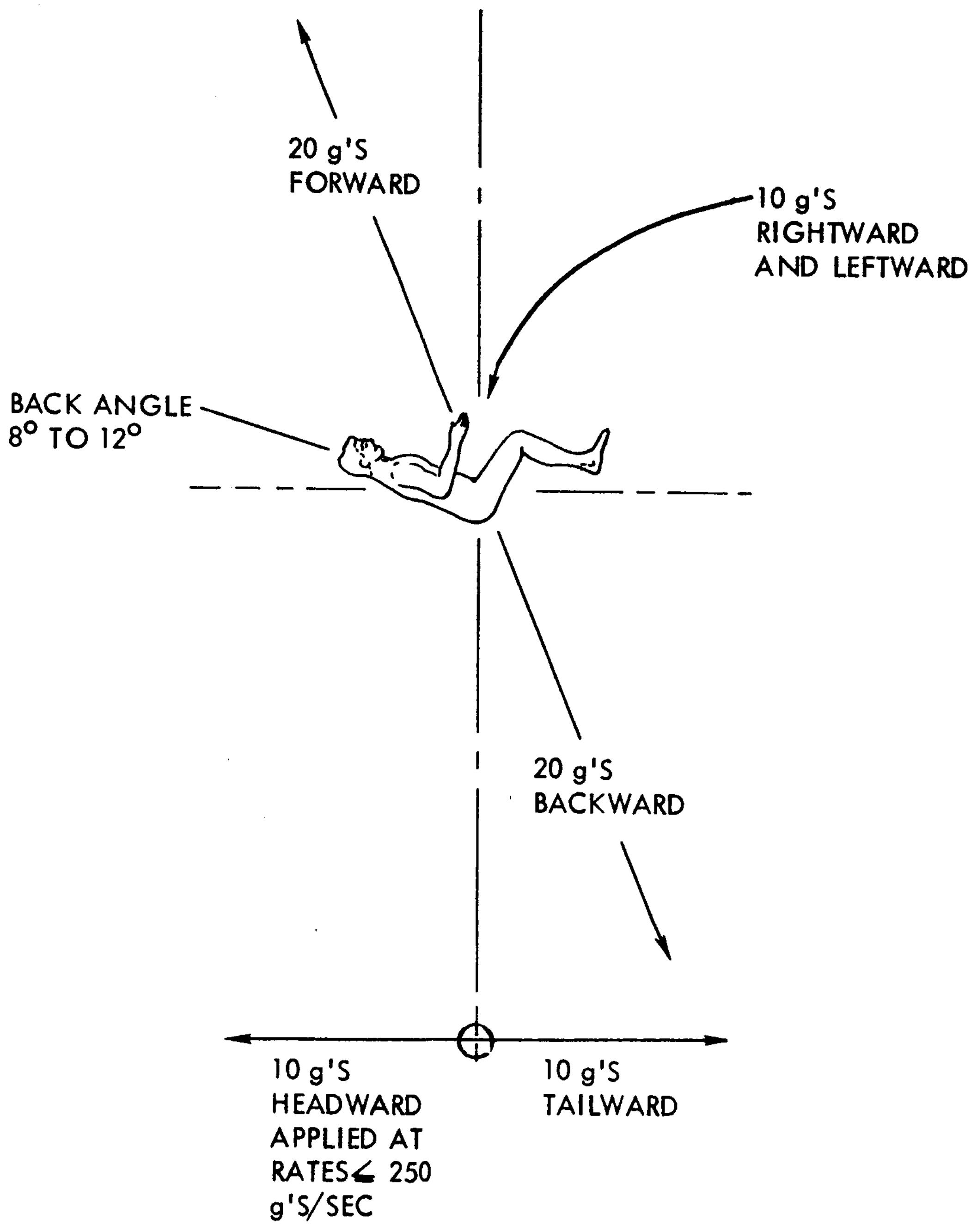

Figure 9. Impact Accelerations - Nominal Limits.

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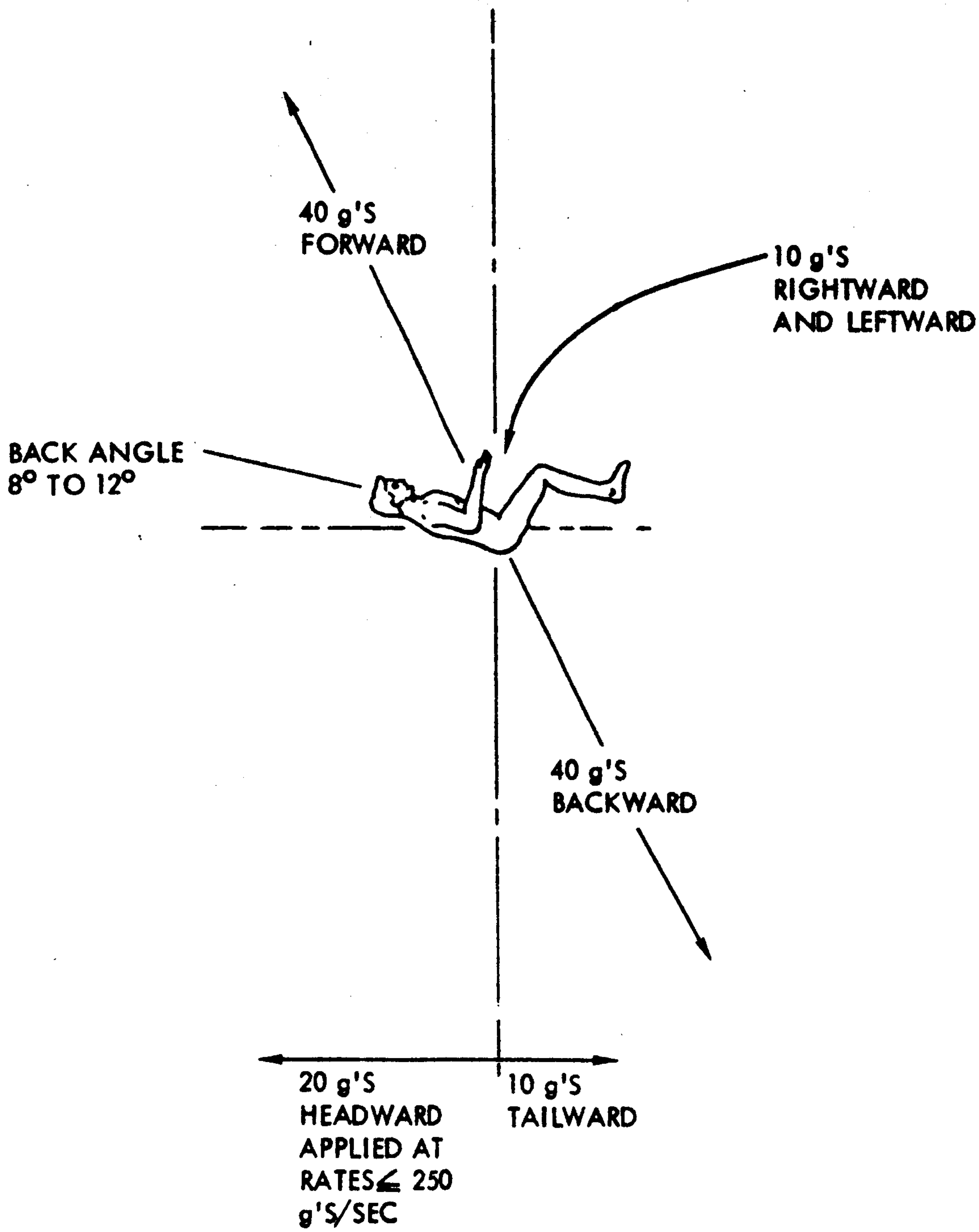

Figure 10. Impact Accelerations - Emergency Limits.

3.3.5.1.2.12.1.9 Parachute Disconnect. - The parachute disconnect shall be assumed to operate within 2 seconds of ground contact.

3.3.5.1.2.12.1.10 Water Landing Impact. - Design requirements shall be as follows:

(a) The Command Module shall not turn over during the dynamics of water contact.

(b) Nominal crew tolerances to impact accelerations and acceleration onset rates shown in Figure 9 shall not be exceeded.

(c) The Command Module primary structure shall remain unimpaired and watertight following the application of dynamic loads during contact.

(d) Accelerations of the Command Module structure, including flexibility effects, shall not exceed 40 g.

A 20-knot steady wind shall be assumed to occur at 10 feet above the mean surface level.

Thermal gradients resulting from sudden water immersion shall be accounted for in the structural design.

3.3.5.1.2.12.1.11 Emergency Landing Impact. - Design conditions within the category of emergency landing impact shall include all possible modes of landing impact of which the aggregate probability of occurrence is greater than P, excepting those covered by normal and water impact. The value of P shall be consistent with the overall probability assigned to emergency limits for crew safety (0.999).

Design criteria to implement this definition cannot be formulated for the initial development phase. However, (1) structural integrity during emergency landing impact shall not be a requirement, and (2) emergency crew tolerances, Figure 10 shall not be exceeded.

3.3.5.1.2.12.1.12 Launch Escape Provisions. - For escape from a catastrophic failure of the Launch Vehicle prior to or shortly after lift-off, the Launch Escape System shall separate the Command Module from the Launch Vehicle and shall propel the Command Module to an altitude of at least 4000 feet and to a lateral range at apogee of at least 3000 feet without exceeding the emergency crew tolerance limits given in Figures 11, 12, 13, and 14. In addition, the Command Module and the Launch Escape System shall withstand the following loading environments acting simultaneously:

(a) Eight psi limit overpressure due to explosion of all or part of the Launch Vehicle.

(b) Sound pressure levels per paragraph 3.3.5.1.2.3.

(c) Vibrations induced by (b).

(d) Mechanically transmitted vibrations due to uneven burning characteristics of the rocket motor.

3.3.5.1.2.12.1.13 Abort at Maximum Dynamic Pressure During Atmospheric Exit. - For escape at maximum dynamic pressure, the Launch Escape System shall separate the Command Module from the Launch Vehicle and shall propel the Command Module to a safe distance from the Launch Vehicle. The Command Module and Launch Escape System combination shall be aerodynamically stable or neutrally stable. The maximum possible Launch Vehicle "miss" distance without exceeding the emergency crew tolerance limits as given in Figures 11, 12, 13, and 14.

Launch Vehicle pitching accelerations of the order of 0.5 rad/sec^2 shall not preclude safe separation and escape.

Primary structures shall be designed for the loads arising from a tumbling of the Command Module-Launch Escape System combination during abort at maximum dynamic pressure during atmospheric exit. Tumbling shall be considered in both the pitch and the yaw planes.

3.3.5.1.3 Design Weights. - The load-carrying structure shall be sized on the basis of the individual module weights. These weights shall not be

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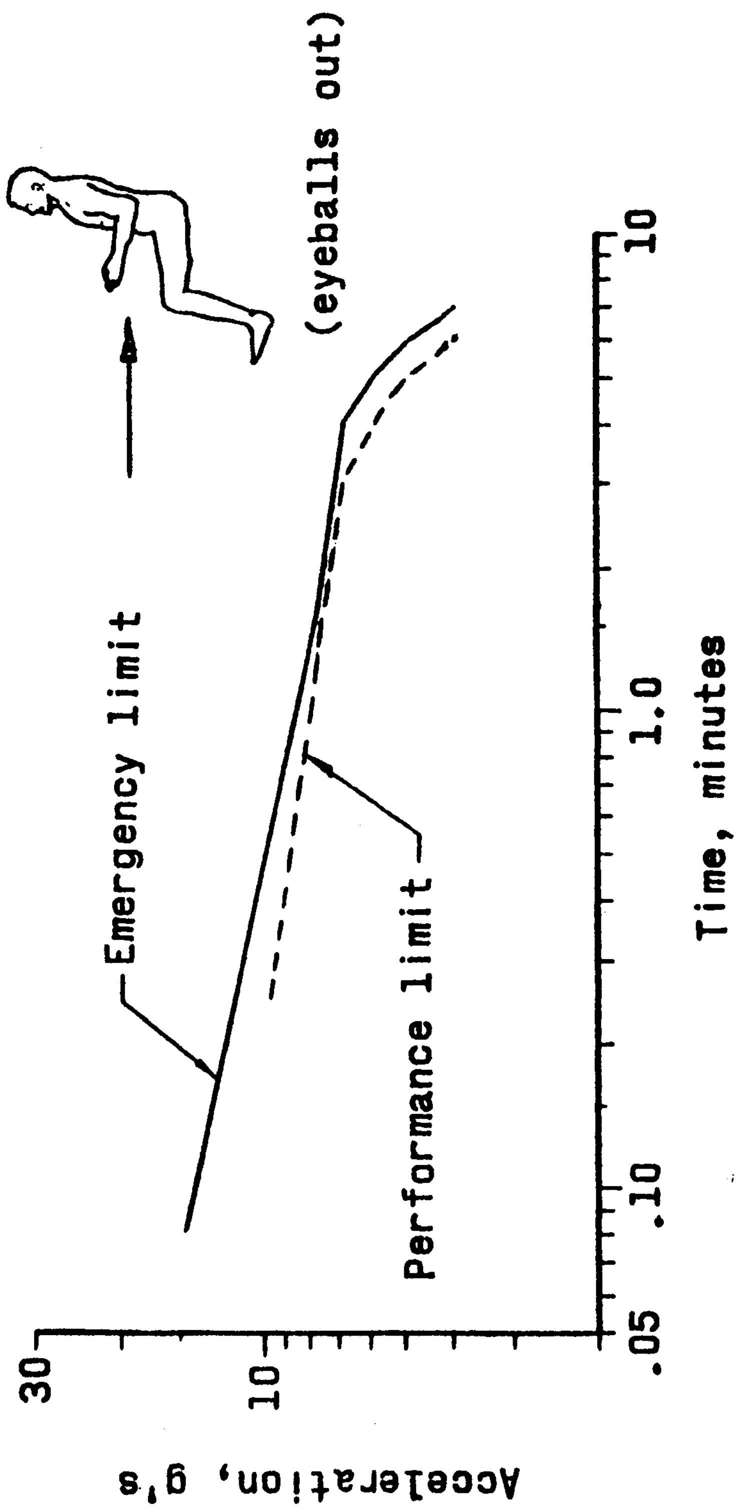

Figure 11. Sustained Acceleration

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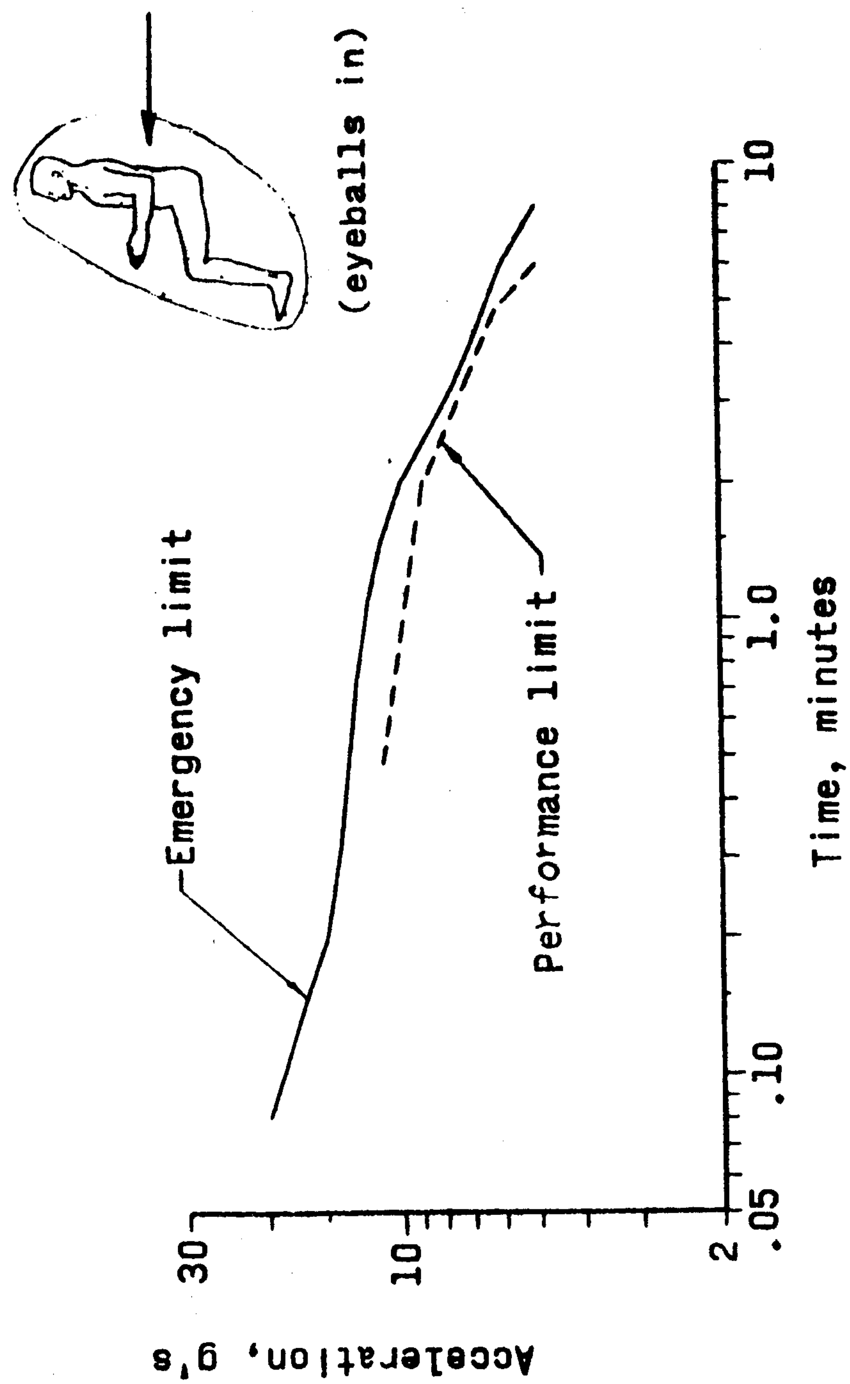

Figure 12. Sustained Acceleration

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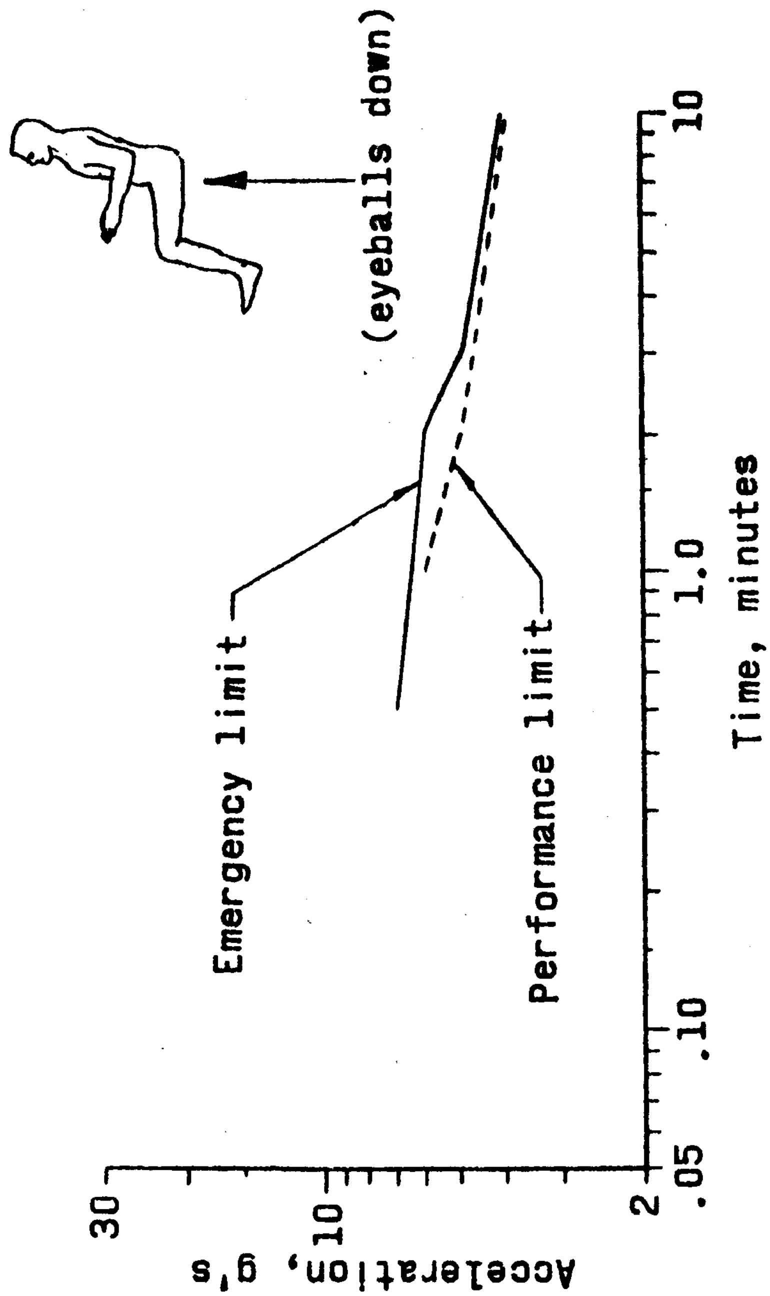

Figure 13. Sustained Acceleration

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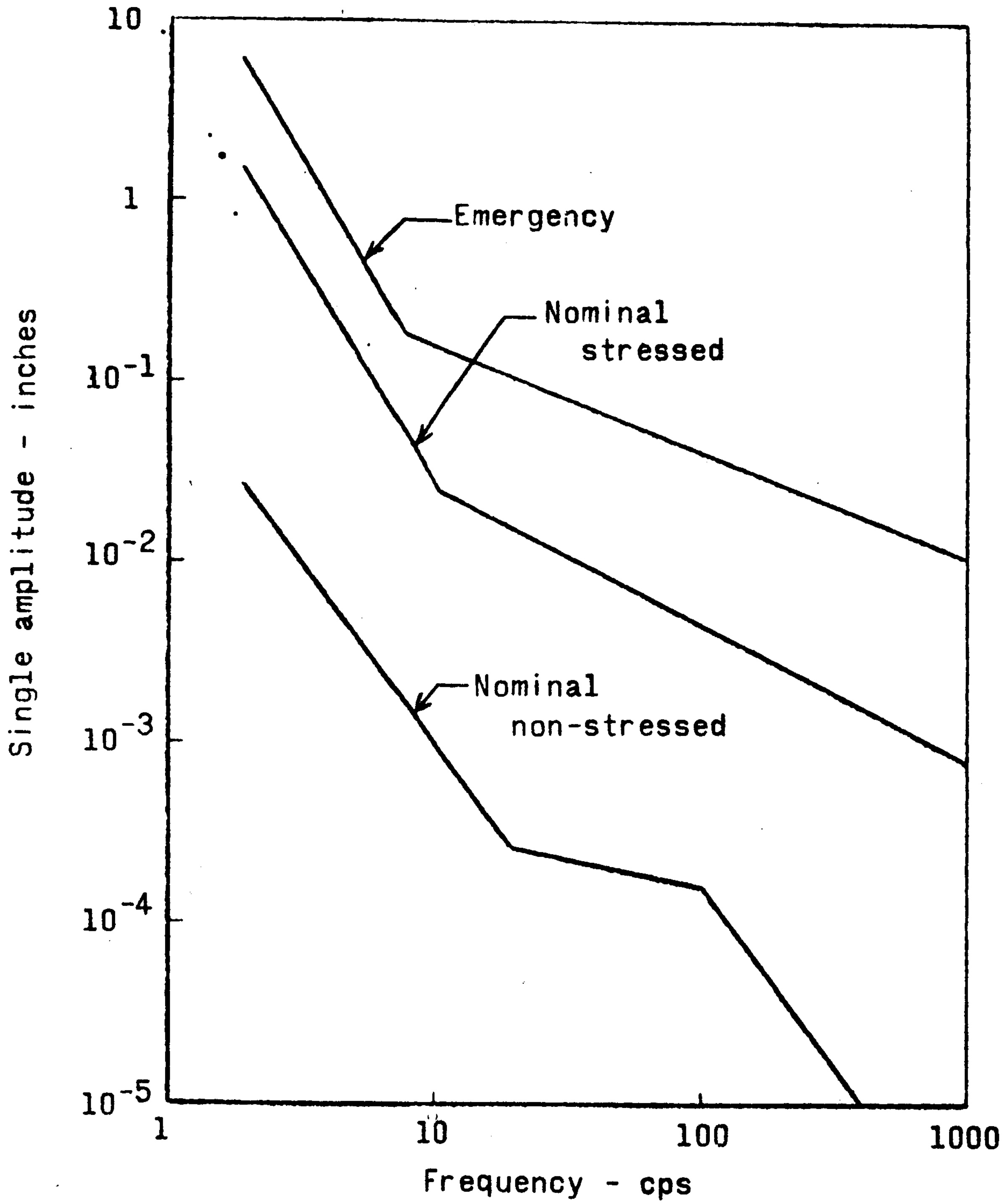

Figure 14. Vibration Limits.

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construed as representing actual module weight goals and every effort shall be made to keep the actual weights to a minimum.

3.3.5.2 Command Module. -

3.3.5.2.1 Heat Shield. - The space between the heat shield and the pressure cabin shall be vented to the outside in a region of low temperature during entry. Penetration and cratering of the heat shield by meteoroids shall not violate the integrity of the heat protection system (See Figure 15).

3.3.5.2.1.1 Material. - Heat shield materials shall be charring-type ablators.

3.3.5.2.1.2 Mechanical Type, Non-Metallic Fasteners. - Fasteners employed to hold the charring ablator to the outer structural shell shall be of a design compatible with the ablative material.

3.3.5.2.1.3 Design Requirements and Considerations. -

(a) Design of charring ablator units shall prescribe the largest unit size possible commensurate with fabrication, producibility, and service requirements.

(b) Design of the charring ablative system shall minimize the existence of heat-leak paths leading to the metallic face of the outer structural shell.

(c) Openings, passages, or orifices through the heat shield system shall be covered by hinged, or otherwise retractable, doors.

(d) Special design effort shall be expended to eliminate possible damage to the charring ablator by attitude control rocket firings.

(e) Design and placement of escape rockets shall be such that damage to the Command Module heat shield system shall not be caused by its firing.

(f) Command Module design shall minimize the loads imposed on the charring ablative material.

(g) During the recovery phase, the forward heat shield shall be jettisoned at 60,000 feet.

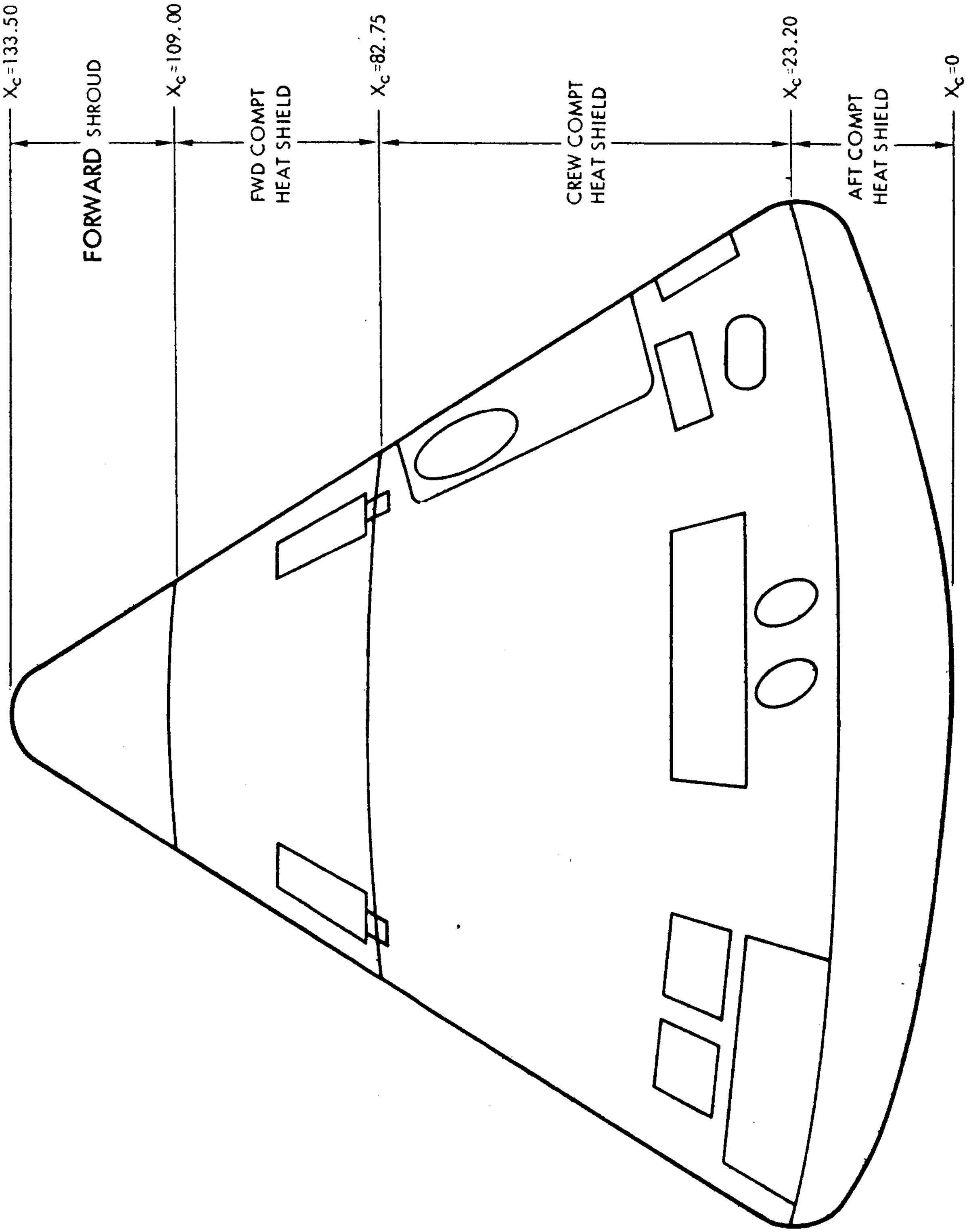

Figure 15. Command Module Heat Shields

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3.3.5.2.2 Outer Structure.- The outer structure shall be the backup structure for the heat shield ablation material and shall carry no primary loads.

3.3.5.2.3 Inner Structure.- The inner structure shall be the primary structure of the Command Module and will be the pressure cabin for the crewmen. The structural integrity of the pressure cabin shall not be violated as a result of normal earth or water landings. The cabin shall have a factor of safety of 1.5.

3.3.5.2.4 Couches, Crew Support and Restraint System.-

3.3.5.2.4.1 Design Approach.- Design of the crew support and restraint systems shall be integrated with the design of the earth landing and launch escape propulsion systems.

3.3.5.2.4.2 Couch.- Each crewman shall be provided with a support couch for protection against acceleration loads. The couch shall provide full body and head support during all nominal and emergency acceleration conditions. Couch construction and materials shall not amplify any accelerating forces by a factor of more than 1.2.

3.3.5.2.4.3 Restraint System.- A restraint system shall be provided with each couch.

3.3.5.2.5 Impact Attenuation.- Impact attenuation beyond that required to maintain general Spacecraft integrity shall be obtained through use of discrete shock mitigation devices for individual crew support and restraint systems.

3.3.5.2.5.1 Vibration Attenuation.- Vibration loads transmitted to the crew shall be kept within tolerance limits that permit the crew to exercise necessary control and monitoring functions.

3.3.5.2.6 Equipment Installation and Bracketry.- All equipment installation and bracketry shall be designed to withstand the accelerations due to vibrations superimposed upon the steady-state accelerations.

3.3.5.2.7 Flutter and Vibrations.- The structure shall be designed to withstand local structural vibrations, and an acoustic attenuation of 30 db through the basic Command Module structure shall be attained.

3.3.5.3 Service Module.-

3.3.5.3.1 Structural Shell.- The structural shell shall provide meteoroid shielding for vulnerable components. Energy absorption levels will be defined at a later date.

3.3.5.3.2 Radiator Panels.- Four radiator panels shall be required, two for ECS cooling and two for EPS cooling.

3.3.6 Stabilization and Control System (SCS).-

3.3.6.1 Function.- The Stabilization and Control System shall provide signals for angular orientation and stabilization of the Spacecraft about three axes under command of the crew, the guidance system, the horizon scanner, the sun sensor or the self-contained standby inertial reference system, and shall provide translational control during rendezvous and docking, and thrust vector control during midcourse corrections.

3.3.6.2 Performance Requirements.- The performance required of the SCS during the various phases of a lunar mission is shown in Table I.

Table I. SCS Performance Requirements

Mission Phase	Separation Thrust	Attitude Control	Translation Control	Thrust Vector Control
Ascent	LES	Booster	Booster	Booster
Parking Orbit	—	Booster	Booster	Booster
Translunar Injection	Booster	Booster	Booster	Booster
Translunar Midcourse	—	S/M	S/M	S/M
Lunar Orbit Injection	—	S/M	—	S/M
Lunar Orbit Maneuvers	—	S/M	S/M	—

Table I. (Cont.)

Mission Phase	Separation Thrust	Attitude Control	Translation Control	Thrust Vector Control
Transearth Injection	—	S/M	—	S/M
Transearth Midcourse	—	S/M	S/M	S/M
Transearth-Entry Interface	S/M	C/M	—	—
Entry	—	C/M	—	—
Recovery	—	C/M	—	—

3.3.6.3 SCS Components. - One each of the following components (except as noted) shall comprise the stabilization and control system.

3.3.6.3.1 Rate Gyro Package. - The rate gyro package shall contain those sensing elements and associated circuitry required to provide angular rate signals in roll, pitch and haw for control and display information as required.

3.3.6.3.2 Inertial Reference Package. - The inertial reference package shall contain those sensing elements and associated circuitry required to provide angular displacement signals, in addition to those derived from the Guidance and Navigation system, necessary for stabilization, control and display required in the stabilization and control system. This device will serve as a backup to the Guidance and Navigation system IMU.

3.3.6.3.3 Mode Select Panel. - The mode select panel shall contain the necessary elements to enable the crew to designate the appropriate SCS operation mode. There shall be two of these subassemblies per SCS.

3.3.6.3.4 SCS Adjust Panel. - The SCS adjust panel shall provide means by which the crew may adjust various SCS parameters such as reaction-jet thrust level, gyro drift trim, system deadband, and attitude-to-rate ratio gain.

3.3.6.3.5 Electronic Control Package. - This device shall contain those circuit elements necessary for resolving, summing, shaping, and switching of these sensor and manual command input signals necessary to control the operation of the Service Module and Command Module reaction jets and the thrust vector subsystem of the Service Module. This device shall also contain the circuit isolation and signal conditioning elements necessary for inflight checkout, monitoring, and telemetry requirements.

3.3.6.3.5.1 Electrical Power Conversion. - The electrical power conversion subassembly shall contain the devices and circuitry necessary to accept the spacecraft electrical power and generate and regulate the necessary voltages and frequencies required for satisfactory performance of the stabilization and control system.

3.3.6.3.6 Display Electronics Package. - This device shall contain the electronics required for use with SCS displays.

3.3.6.3.7 Accelerometer Package. - This package shall provide acceleration data from body-mounted accelerometers for midcourse ΔV display and backup reentry information.

3.3.6.3.8 Flight Director Indicator. - This device shall display vehicle attitude and angular rate information for crew control and monitoring purposes. In addition to displaying actual attitudes, the indicator shall show command attitudes as inserted by the crew or the guidance and navigation system. There shall be two of these subassemblies per SCS.

3.3.6.3.9 Flight Director Control Panel. - This device shall provide controls for crew selection of appropriate attitude reference and mission modes. There shall be two of these subassemblies per SCS.

3.3.6.3.10 Horizon Scanner. - This device shall sense the location of both the terrestrial and lunar horizon in the lateral and longitudinal planes with respect to the vehicle body reference axes during orbital phases of flight.

3.3.6.3.11 Manual Translational Controls. - These controls are crew actuated translation controls necessary to control translation of the vehicle in any of six directions. There shall be two of these subassemblies per SCS.

3.3.6.3.12 Manual Two-Axis Rotational Controls. - These controls are crew actuated controls necessary to manually control rotation of the vehicle about the pitch and roll axes. There shall be two of these subassemblies per SCS.

3.3.6.3.13 Manual Yaw Rotational Control. -1 These controls are crew actuated controls necessary to manually control rotation of the vehicle about the yaw axis. There shall be two pairs of these per SCS.

3.3.6.3.14 Manual Thrust Control. - These controls are crew actuated controls necessary to provide control of the Service Module. There shall be two of these subassemblies per SCS.

3.3.6.3.15 ΔV Indicator. - The ΔV indicator shall display actual ΔV magnitude and "time-to-go to ΔV" for pilot monitoring and control purposes. The ΔV indicator shall also provide the means for manually inserting the desired ΔV magnitude and the desired "time-to-go to ΔV" into the SCS ΔV control subsystem. There shall be two of these indicators per SCS.

3.3.6.3.16 Translational Velocity Indicator. - This device shall display translational velocity information relative to the Y and Z body axes for use by the pilot in monitoring and controlling the spacecraft translational velocity. There shall be two of these per SCS.

3.3.6.4 Weight. - The total weight of the equipment shall be the minimum consistent with other design requirements and good design practice.

3.3.6.5 System Manual Controls. - The SCS manual controls shall provide maximum utilization of system capabilities. It shall be possible to select automatic modes for those operations beyond human capability or to relieve the astronauts of continuing tedious tasks. The manual controls shall have the following common characteristics:

(a) Natural correspondence of motion between the command and the required reaction.
(b) Suitable "feel" characteristics.
(c) Dual controls, wherever specified, shall be paralleled, with equal authority available to each set.

3.3.6.5.3 Pitch and Roll Rotation. - Dual pitch and roll rotation controls shall be provided. A means shall be provided which shall permit direct emergency operation of the gimballed engines in the event that normal system operation is inadequate.

3.3.6.5.4 Yaw Rotation. - Dual yaw rotational controls shall be provided. A firm detent shall be provided which shall permit direct (emergency) operation of the gimballed engines in the event that normal system operation is inadequate.

3.3.6.5.5 Translation. - Dual translational motion controls shall be provided which shall permit direct (emergency) operation of the reaction control engines in the event that system operation is inadequate.

3.3.6.5.6 Thrust. - Dual thrust controls for the main propulsion engine shall be provided.

3.3.6.6 Engine Utilization. - Engine utilization shall consist of:

(a) Attitude control

(b) Translation control

(c) Rate stabilization

(d) Large velocity corrections

3.4 Modules. - The Command Module, Service Module, and Adapter described in the following paragraphs represent a partial Spacecraft. As additional modules are incorporated to formulate a Lunar Landing Capability, this specification shall be expanded accordingly.

3.4.1 Command Module. -

3.4.1.1 Function. - The Command Module shall be designed to provide a habitable environment for three human beings during all phases of Project Apollo spaceflight for a maximum of 2 weeks duration. The Command Module shall remain essentially unchanged for all Apollo Missions. All crew-initiated control functions shall be exercised from the Command Module. The Command Module shall be the command and control center of the Spacecraft.

3.4.1.2 Geometric Parameters. -

3.4.1.2.1 Dimensions. - The basic external geometry of the Command Module based on aerodynamic and thermodynamic performance requirements shall be:

Overall height = 133.5 ± 0.5 inches

Diameter = 154 ± 0.5 inches

Structure outline = See Figure 16

3.4.1.2.2 Weight. - The following weight shall be a design objective:

Gross Take-off Weight = 8,500 lbs.

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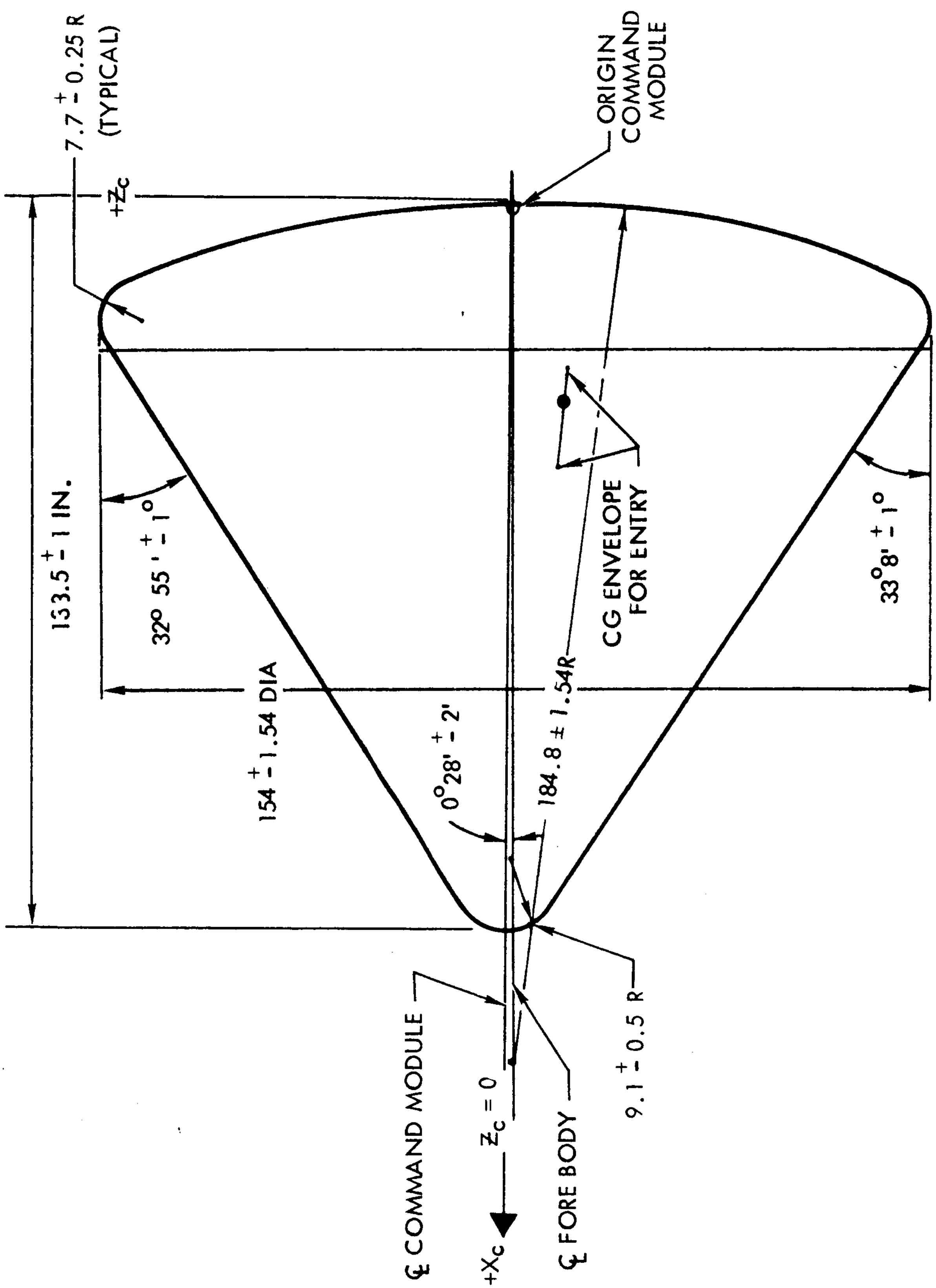

Figure 16. Command Module

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3.4.1.3 Performance. - The Command Module performance shall include the capability of carrying three crew members, their equipment, and the required systems in an inhabitable environment throughout all phases of the Apollo mission except to provide ingress and egress while in space, with cabin decompression.

3.4.1.3.1 Command Module Systems. - The performance characteristics of the Command Module shall be defined by the performance capabilities of the following systems as applied to the Command Module:

1. Structural
2. Crew
3. Environmental Control
4. Launch Escape
5. Reaction Control
6. Guidance and Navigation
7. Earth Landing
8. Electrical Power
9. Communications and Instrumentation

3.4.1.3.1.1 Structural System. -

3.4.1.3.1.1.1 Function. - The function of the Command Module structural system shall be to sustain normal ground and flight loads, support maximum abort and landing impact loads, provide a mounting surface for all Command Module systems, provide a vessel for presurization, decrease the flux density within the capsule due to radiation, provide protection against the damaging effects of meteoroids, provide crew living quarters during various mission phases and provide thermal protection during the boost and entry phases of the flight. Additional structural requirements are given in paragraph 3.3.5.

3.4.1.3.1.1.2 Aerodynamics. - The module shall be a blunt body developing a hypersonic L/D ratio of approximately 0.50. The L/D vector shall be effectively modulated in hypersonic flight by roll control.

3.4.1.3.1.1.3 CG Management. - The Command Module shall be designed for equipment location such that the Command Module CG shall be optimized for the maximal L/D ratio and maneuverability during the entry phase.

3.4.1.3.1.1.4 Inboard Profile. - The internal arrangement of the Command Module shall contain three compartments: crew, forward, and aft. This arrangement is illustrated in Figures 17 through 23.

3.4.1.3.1.1.4.1 Crew Compartment. - The crew compartment shall house the crew and required systems. Equipment and storage space shall be contained in the following locations:

(a) Lower Equipment Bay
(b) Upper Equipment Bay
(c) Right-Hand Equipment Bay
(d) Left-Hand Equipment Bay

3.4.1.3.1.1.4.1.1 Lower Equipment Bay. - The lower equipment bay shall have storage cabinets and drawers with recessed draw pull latches. The sextant shall be located in the upper center of the bay with food storage cabinets directly below. The Inertial Measurement Unit (IMU) equipment shall be located in the vicinity of the food storage area. Electronics telecommunications equipment shall be stored below the food storage area. There shall also be provisions for scientific equipment storage and tools on the left side of the lower equipment bay. Waste storage area shall be on the lower right and the areas between tool storage and waste are as yet undefined.

3.4.1.3.1.1.4.1.2 Upper Equipment Bay. - The upper equipment bay shall be considered as a fitted storage area for pressure suits, backpacks, helmets, and survival equipment.

3.4.1.3.1.1.4.1.3 Right-Hand Equipment Bay. - The right-hand equipment bay shall contain storage cabinets. The upper left area shall contain electronic modules and spares. The lower left area shall contain storage cabinets with recessed inner pull latches for tools, medical and surgical supplies, toiletries, and hygiene supplies. The lower right-hand area shall contain cabinets for cleansing pads and clothes storage.

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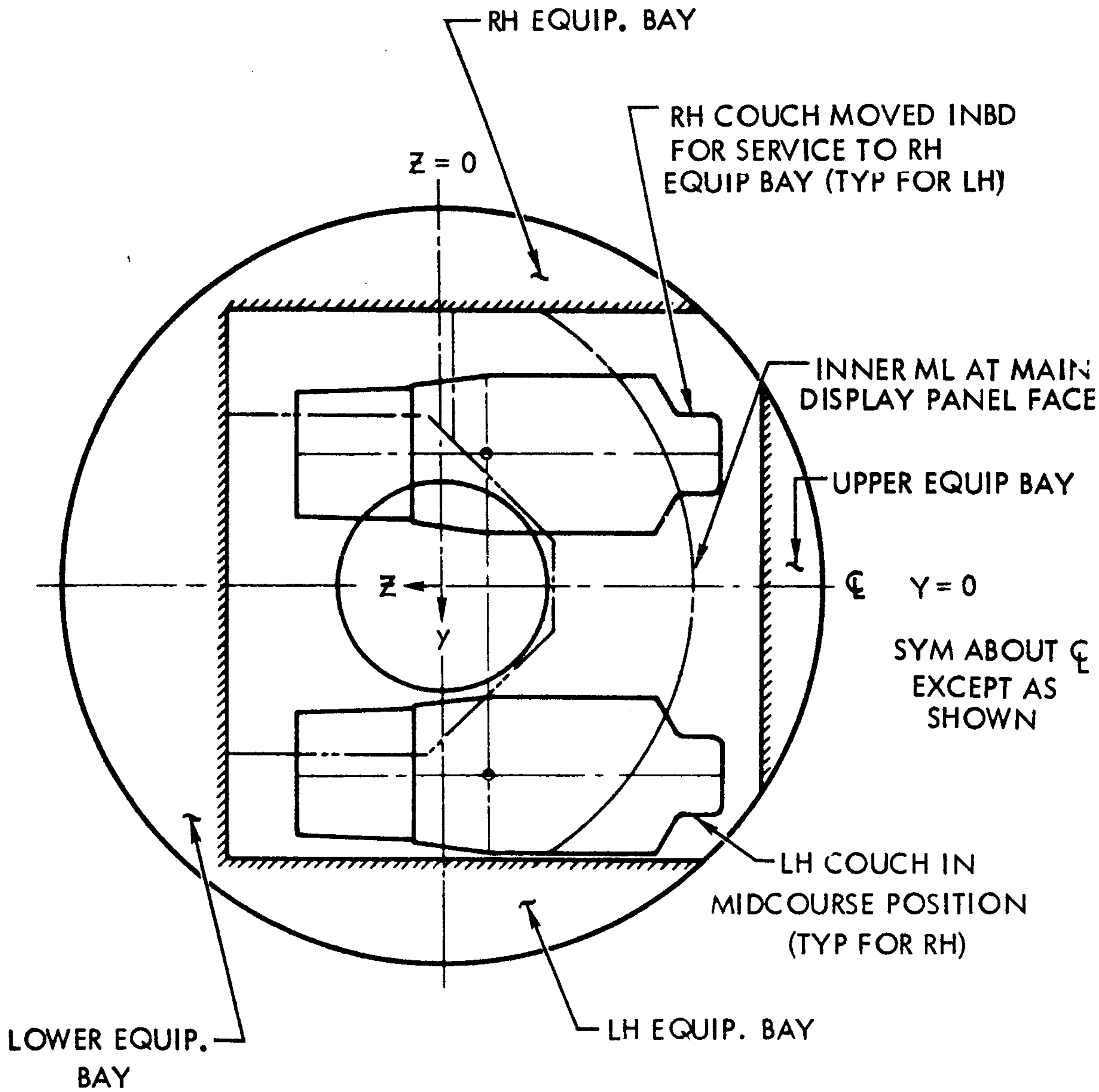

Figure 17. Area Designations

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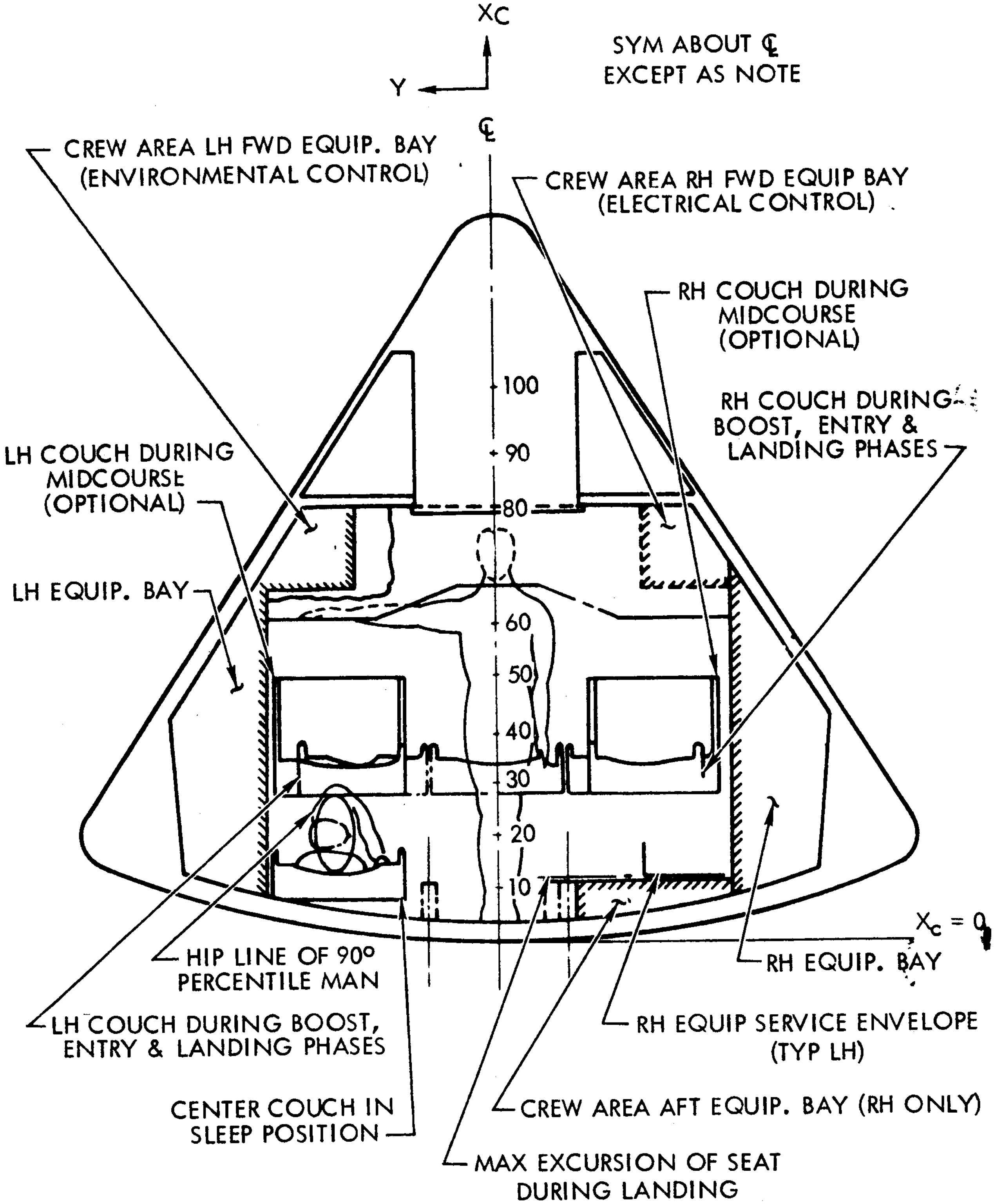

VIEW LOOKING TOWARD LOWER END OF CREW AREA

Figure 18. Area Designations

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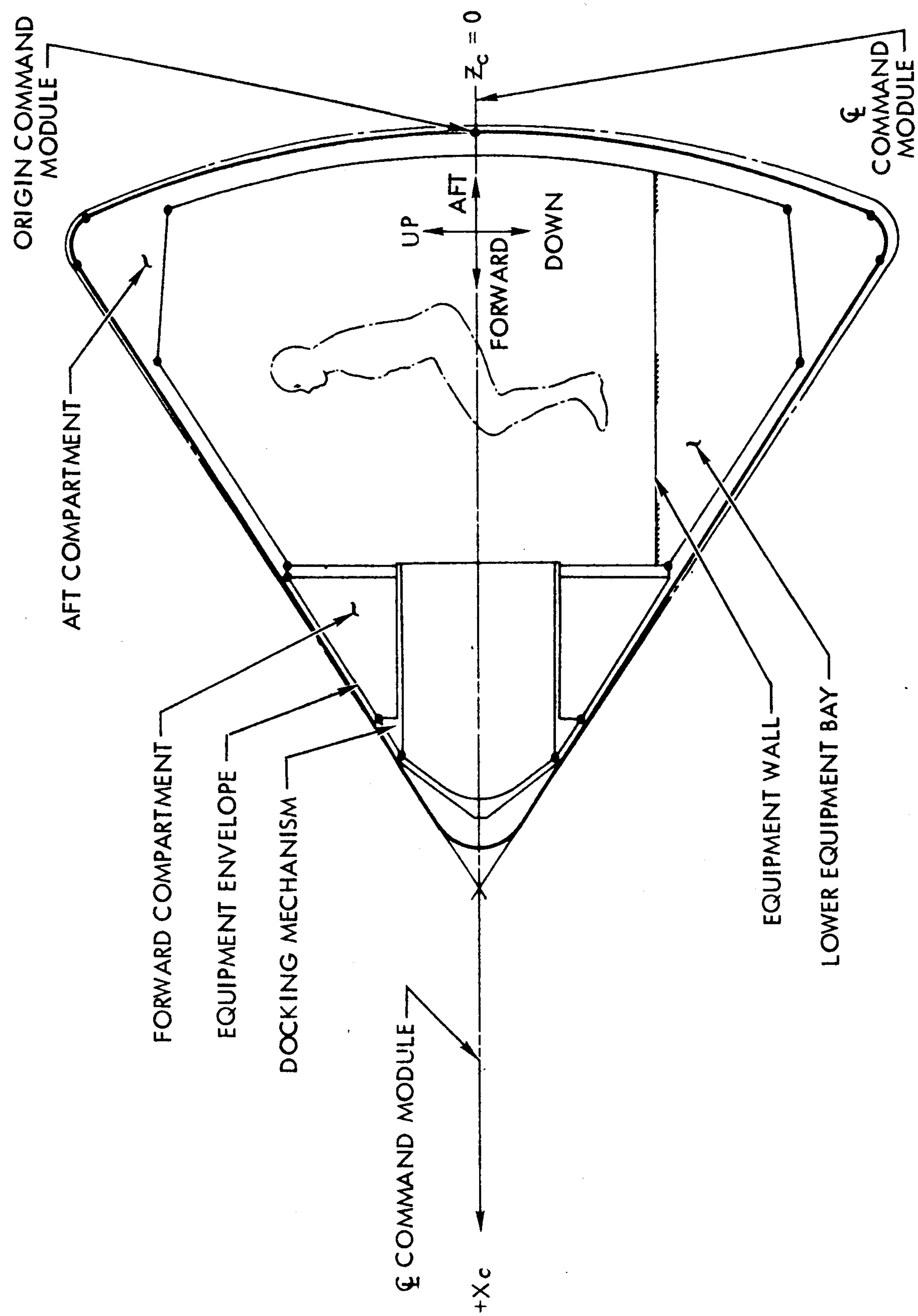

Figure 19 AREA DESIGNATIONS SIDE VIEW

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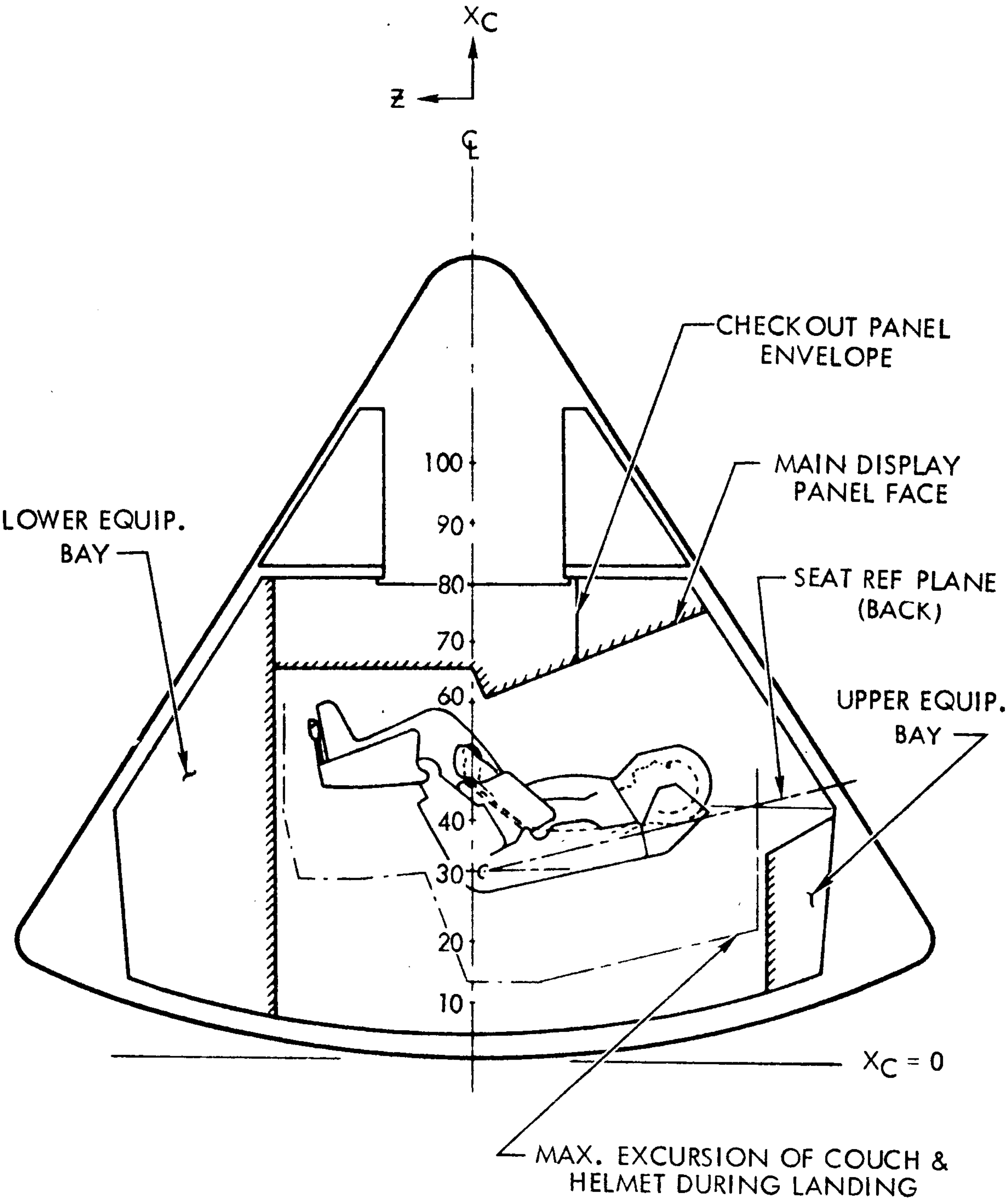

Figure 20 Area Designations

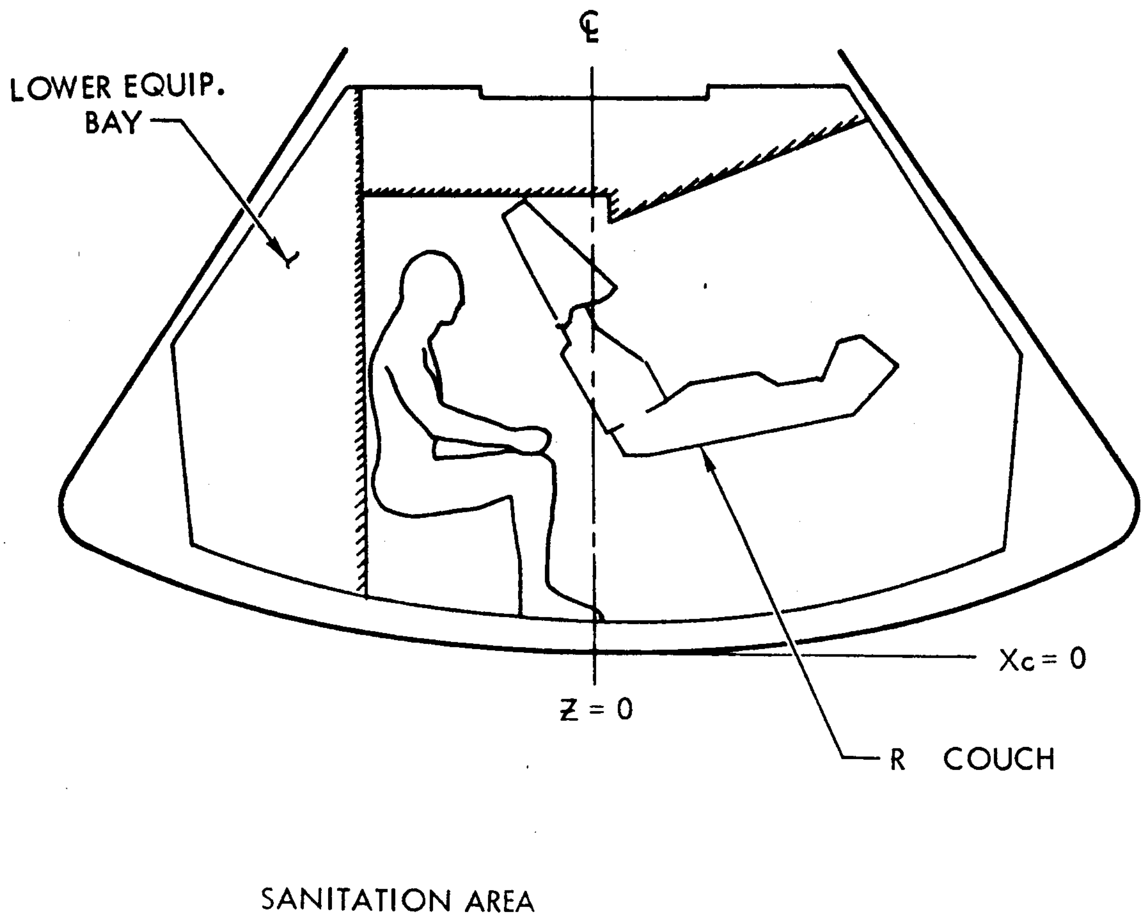

Figure 21 Area Designations

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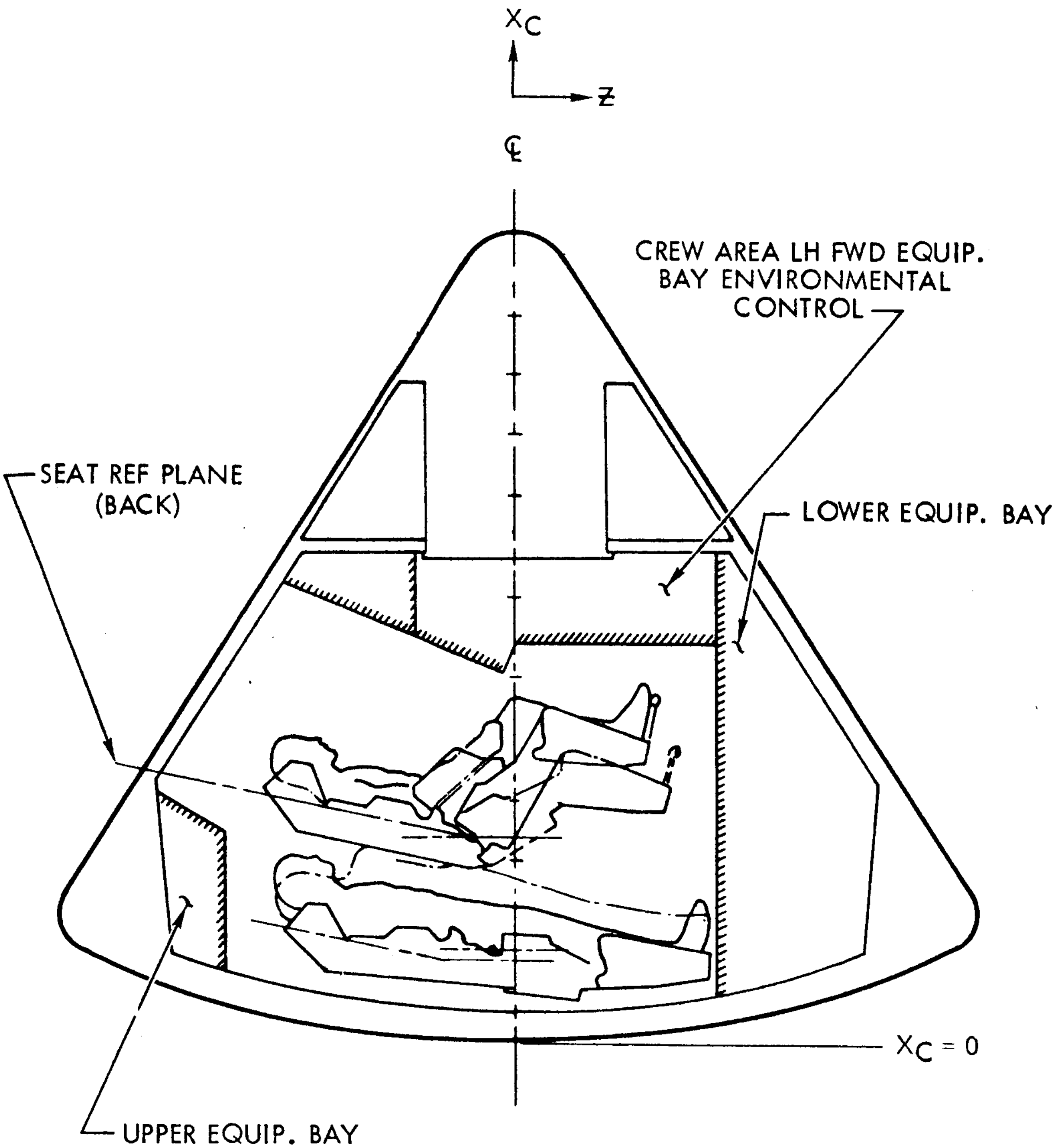

Figure 22 Area Designations

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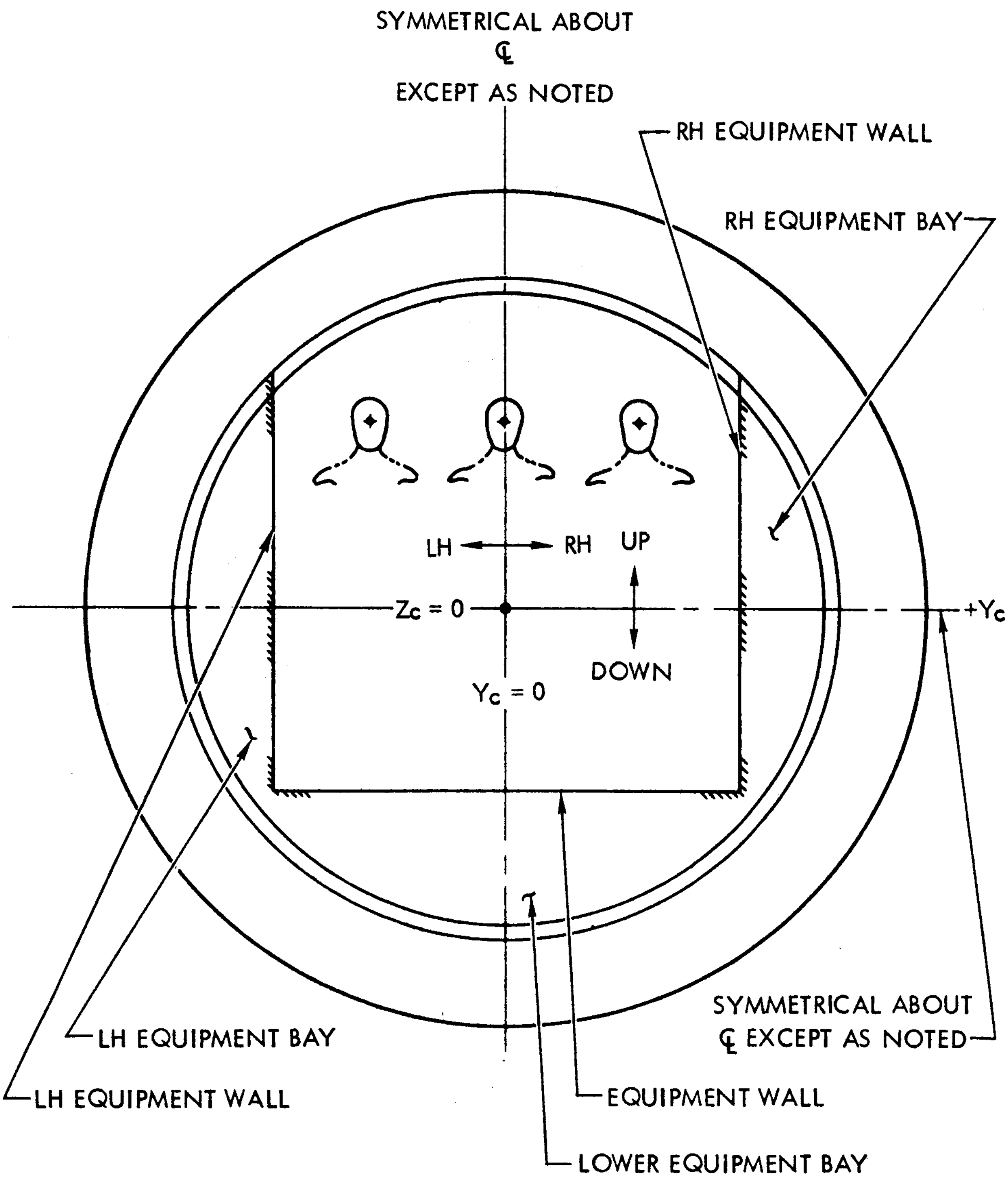

VIEW
LOOKING FORWARD

Figure 23 Area Designations

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3.4.1.3.1.1.4.1.1.4 <u>Left-Hand Equipment Bay</u>. - The left-hand equipment bay shall contain the absorption cylinders. Above the cylinders shall be the remaining components of the environmental control system. The remainder of the bay shall be available for spare parts and instrument storage.

3.4.1.3.1.1.4.2 <u>Forward Compartment</u>.- The forward compartment shall be located in an area forward of the crew compartment. This compartment shall house components of the earth landing system, reaction control system, and antennas for communication-instrumentation system.

3.4.1.3.1.1.4.3 <u>Aft Compartment</u>.- The aft compartment shall be located on the aft periphery of the command module. This compartment shall house components of the reaction control system, environmental control system, earth landing system, separation system, crew system, and umbilical connections.

3.4.1.3.1.1.5 <u>Ingress and Egress</u>.-

3.4.1.3.1.1.5.1 <u>Central Hatch</u>.- Normal pre-launch ingress to the Command Module for the crew shall be through a central hatch.

3.4.1.3.1.1.5.2 <u>Forward Hatch</u>.- The Command Module shall incorporate a forward hatch for crew ingress and egress to and from the Command Module in the environment of space.

3.4.1.3.1.1.5.3 <u>Emergency</u>.- A single blowout panel shall provide the crew with bailout or other types of emergency egress. The panel shall be designed to jettison outward by detonating a linear explosive charge. The explosive charge shall be detonated by actuating a switch located on the left hand console.

3.4.1.3.1.1.6 <u>Windows</u>.- The Command Module shall have a minimum of five windows to aid in providing maximum use of direct vision during rendezvous, earth landing, and scientific observations, for monitoring crewmen outside the Spacecraft, and for general orientation. These windows shall consist of multiple panes and shall be protected by covers during launch and entry. The covers shall be operated from inside the cabin by the crew members.

3.4.1.3.1.1.7 <u>Mirrors</u>.- Mirrors shall be provided to increase the external visual field. The number and location of mirrors shall be determined by design concepts of window covers.

3.4.1.3.1.1.8 <u>Attachments</u>.- Provisions shall be incorporated into the Command Module design for attachment of ground handling equipment, the launch escape system, and the Service Module.

3.4.1.3.1.1.9 <u>Design Leak Rate</u>.- The maximum design leak rate of the Command Module shall be 0.2 lbs/hr.

3.4.1.3.1.1.10 <u>Flotation</u>.- The structural design shall incorporate a flotation capability which shall not be impaired by earth or water impact.

3.4.1.3.1.2 Crew System.- The crew system shall support needs peculiar to the presence of human beings aboard the Spacecraft for a period of 14 days. Crew systems shall involve that portion of the design related to the adequate functioning of the 3 crewmen. The crew system shall involve the man-equipment relationships defined in paragraphs 3.4.1.3.1.2.1 through 3.4.1.3.1.2.7.

3.4.1.3.1.2.1 Controls and Displays.- Command Module controls and displays shall be provided as required to permit the crew to control the Spacecraft and monitor the parameters considered critical to an acceptable mission performance. Controls and displays shall include the following organization of component panel display and control divisions, sections, and areas. The arrangement and physical characteristics of the controls and displays shall be as follows:

- Primary Panel Arrangement
  - Control Station
  - Center Station
  - Systems Management Station
- Armrest Controls
- Left Hand Console
- Right Hand Console
- In-Flight Test Panels
- Cabin Oxygen Outlets

3.4.1.3.1.2.1.1 Primary Panel Arrangement.- The Primary Panel arrangement shall consist of three station panels.

3.4.1.3.1.2.1.1.1 Control Station Panel.- The control station shall be positioned above the control station operator's couch as oriented in the launch position. The panel shall contain the following indicators and controls:

- Trajectory Error
- Trajectory Correction
- Optical Display
- Propulsion Control
- Attitude Control
- Range-Velocity Elapsed Time Indicator
- Entry and Abort Display
- Time Display
- Warning Lights
- SCS Mode Select

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Flight Director Control

Master Caution Light

Altitude Rate Indicator

Television

Airspeed

Altimeter

Flight Director Indicator - Attitude Indicator

Rate-Range Indicator

Radar Rendezvous

Camera

Angular Rate Indicator

Back-Up Survival Indicator

Master Caution

Advisory Lights

Docking System

3.4.1.3.1.2.1.1.2 <u>Center Station Panel</u>.- The center station panel shall be located in the center of the primary panel arrangement. The panel shall contain the following indicators and controls:

Audio Control

Flight Director Control

Master Caution Light

Service Module Propellants

Survival Indicator

Flight Director Indicator

Rate-Range Indicator

Entry and Abort Display

Angular Rate Indicator

Camera Controls

Master Caution Controls

3.4.1.3.1.2.1.1.3 Systems Management Station Panel.- The systems management panel shall be positioned above the systems management station as oriented in the launch position.

3.4.1.3.1.2.1.2 Armrest Controls.- Removable control grips shall be provided on the right armrests of the left and center couches to provide pitch, roll, and yaw attitude control. Yaw control shall be provided by foot pedals built into the outboard couches. Removable control grips shall be provided on the left armrests of the left and center couches to provide translation and propulsion thrust magnitude control.

3.4.1.3.1.2.1.3 Left-Hand Console.- The left-hand console shall be located to the left of the primary display panel and shall consist of the following controls and displays:

Lighting Controls

Egress Hatch Firing Switch

Audio

Earth Landing and Recovery Controls

Emergency Launch Escape Controls

3.4.1.3.1.2.1.4 Right-Hand Console.- The right-hand console shall be located to the right of the primary control display panel. This control panel shall consist of the following controls and displays:

Radiation Displays

Audio Controls

Antenna Controls

Printer

3.4.1.3.1.2.1.5 In-Flight Test Panel.- The in-flight test panel shall be located at the base of the primary panel arrangement and shall contain the controls and indicators to test for system malfunctions. This panel shall support the IFTS concept defined in paragraph 3. 3. 4.

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3.4.1.3.1.2.2 Communications and Aural Equipment.- Two-way voice communication capability between individual crew members, between the Spacecraft and earth-based stations, and between each Spacecraft in a rendezvous maneuver shall be provided. Audio controls shall consist of off-transmit-receiver, VHF/DSIF, and intercom controls. Thumb wheels shall be provided for frequency selection. Each crewman shall have his own audio control panel to perform as described in paragraph 3.3.1.2.8.4.

3.4.1.3.1.2.3 Lighting System.- The Command Module shall be illuminated by indirect floodlighting of the primary and secondary duty station panels and general area. Integral illumination shall be incorporated in the displays monitored or used throughout the mission. All interior light and light sources shall be controlled to provide a totally integrated visual environment.

3.4.1.3.1.2.3.1 Indirect Illumination.- Indirect illumination shall consist of multiple lamps located to provide illumination with a minimum of glare. The indirect lighting shall be capable of intensity variation ranging from off to full system capacity. Control shall be provided for the functional crew station areas so that each area may be illuminated to the levels required.

3.4.1.3.1.2.3.2 Integral Illumination.- Integral illumination shall consist of light sources located within the display. This illumination shall provide the required contrast between indicia and background during periods when low luminance levels are required. This illumination shall be capable of intensity variation from off to full system capacity.

3.4.1.3.1.2.3.3 Portable Light.- Independent portable lights shall be provided in the Command Module to permit the crew members to view areas or equipment not normally lighted.

3.4.1.3.1.2.3.4 Window Filters.- Window filters shall be provided to control exterior illumination with respect to the Command Module interior lighting.

3.4.1.3.1.2.3.5 Rest Area Light.- A small light shall be provided in the rest station to permit the off-duty crew member to read or write as necessary.

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3.4.1.3.1.2.4 Couches. - Couches shall be capable of being oriented to provide restraint and comfortable support during all mission phases. The seat shall be approximately 24 inches wide with a slight lateral curvature which will accommodate a crew man in a pressurized suit. The couch shall be covered with a "slow memory" liner to improve load distribution and absorb shock on landing impact. A compact liner shall overlay the slow memory liner.

3.4.1.3.1.2.4.1 Restraint System. - The restraint system shall provide restraint for crew members during launch and entry and during zero or low g conditions. The restraint system shall consist of parachute-harness-type-webbing restraint belts attached to the pressure suit at shoulders and hips and shall be compatible with the restraint attach points of the couch. The belt fasteners shall be the quick disconnect type. The belts and fasteners shall provide for restraint to 40 g.

3.4.1.3.1.2.4.2 Foot Restraint. - Toe cups shall provide restraint for the feet.

3.4.1.3.1.2.4.3 Arm Restraint. - Arm restraint shall be provided by the couches.

3.4.1.3.1.2.4.4 Hand Grips. - A hand grip shall be provided along the lower edge of the instrument panel.

3.4.1.3.1.2.4.5 Roll, Pitch and Yaw Controls. - Roll, pitch and yaw controls shall be incorporated in the left hand and center couch arm rests.

3.4.1.3.1.2.4.6 Adjustable Areas. - The following couch areas shall be adjustable:

Headrest

Backrest

Hip and knee angle - 90 degrees to 180 degrees with the normal launch and entry angle 108 degrees.

Armrest - adjustable for 90 degrees to 180 degrees relative to the backrest.

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3.4.1.3.1.2.4.7 Sleeping Facilities. - The center couch shall be capable of being repositioned for use as a sleeping area. It shall be relocated on the floor beneath the control station couch and shall accommodate one crew member. An adjustable cover, attached along three edges of the couch and the wall shall isolate the area. A two-way stretch, open-mesh coverlet when drawn over the crew man, shall hold him in place against an inflated mattress. The isolation cover shall close out cabin light and reduce noise level. A manually operated light shall be available within the compartment, and air shall be circulated at a flow rate (manually adjustable) between 3 and 12 cubic feet per minute.

3.4.1.3.1.2.4.8 Couch Environmental Control Fittings. - Environmental control fittings shall be mounted on each couch and shall be compatible with the environmental control fittings on the pressure suits.

3.4.1.3.1.2.5 Exercise and Recreation Equipment (NASA Supplied). - Exercise and recreation equipment shall be provided for the crew.

3.4.1.3.1.2.6 Food Management. - Food management shall include storage and preparation of food and disposal of waste food in bags, and operation of a heating probe. These operations shall be accomplished in the lower equipment bay, center section. Polyethylene film retainers shall cover the individually packaged food products to prevent them from floating out of open drawers. Reconstitution shall be accomplished with hot water injected by probe into the individual containers. Empty bags shall be compressed and stored in the waste disposal area of the lower equipment bay.

3.4.1.3.1.2.7 Personal Hygiene and Bio-Medical Equipment. -

3.4.1.3.1.2.7.1 Oral Hygiene Supplies. - Oral hygiene supplies shall be designed to provide adequate cleansing of teeth and gums by means of a multiple-use toothbrush and ingestible dentrifice. Supplies shall be sealed in individual containers for each crew member. A technique of brushing shall be followed so as to allow no particulate matter to escape into the module atmosphere. After use of equipment, the individual containers shall be reutilized for general storage. Containers shall be air-tight to prevent spread of contamination.

3.4.1.3.1.2.7.2 Razor. - To provide well-being and general cleanliness, a completely portable, battery-operated, vacuum filtered razor shall be provided. The razor shall be designed to hold all hair fragments obtained from 3 men for 14 days. Storage of the razor shall be in the same drawer as oral hygiene equipment, with a dividing partition and separate transparent, retractable covers.

3.4.1.3.1.2.7.3 Toiletries (Defecation-emesis bags, Toilet Tissue). - The defecation-emesis bags shall be designed to provide adequate storage facilities for human waste material. The bags shall be interchangeable for defecation or emesis purposes. A maximum of 40 bags with an estimated packaged volume of 70 cubic inches shall be provided. Toilet tissue also shall be provided in the drawer in addition to the absorbent tissue incorporated into bag design. Use of defecation-emesis bags shall minimize escape of odors and provide for complete sealing of bags to prevent contamination during storage.

3.4.1.3.1.2.7.4 Cleansing and Deodorizer Pads. - Cleansing pads shall be provided for all-purpose washing needs. Individual pads shall be impregnated with lotion cleanser, disinfectant, and deodorizer to satisfactorily meet sanitation requirements. Individual cleansing pads shall be wrapped in multi-use wrappers. Wrapping material shall serve to preserve pads before use, act as a towel or wiping cloth during use, and act as a disposable container for used cleansing pads. Volume requirements of each pad shall be at least 3 cubic inches and 8 pads used per man per day. A volume of 150 cubic inches shall be allowed for storage purposes.

3.4.1.3.1.2.7.5 Blood Pressure Cuff (NASA Supplied). - A standard clinical blood pressure cuff with aneroid (non-mercury) pressure sensor shall be provided. The cuff shall be designed to be used in measurement of arm systolic and diastolic blood pressures in conjunction with the stethoscope and shall occupy approximately 96 cubic inches. The apparatus shall be easily operated by securing the Velcro-type cuff around the subject's arm and squeezing the bulb attached to the cuff until the desired pressure is reached.

3.4.1.3.1.2.7.6 Stethoscope (NASA Supplied). - The stethoscope shall be a standard clinical model and shall have both a bell and diaphragm head; the bell shall be employed in listening for low frequency sounds and the diaphragm for high frequency sounds in chest, heart, and abdomen. The stethoscope shall be operated by engaging the ear pieces snugly in the ears and placing the desired part of the head (bell or diaphragm) lightly over the body cavity to which one is listening. Additional bio-medical equipment is defined in paragraph 3.3.1.2.4.2.

3.4.1.3.1.2.7.7 Orally Administered Drugs. - The orally administered drugs shall be supplied in pill form to be mixed with water in a spare drinking water bag just before taking. They shall include drugs to combat pain, nausea, fatigue, allergic reactions, and infections. Water purification tablets and tranquilizers shall be included. These drugs shall have a slow onset of action.

3.4.1.3.1.2.7.8 Injectible Drugs. - Injectible drugs shall include the same array as the oral drugs with the addition of drugs to combat shock. Syringes shall be disposable.

3.4.1.3.1.2.7.9 Antiseptics. - Antiseptics shall be provided in individual sealed absorbent cotton pad form similar to those used for deodorizer pads to prevent spillage of antiseptic. Individual pads shall be 3 cubic inches in volume and shall be used for cleansing around abrasions, lacerations, and contusions. After use each pad shall be replaced in its sealed bag and stored in the waste disposal area.

3.4.1.3.1.2.7.10 Surgical Supplies. - Surgical supplies shall include gauze dressings, a small Velcro cuff tourniquet, a scalpel, a self-retaining retractor, suture with swedged-on cutting needle, a tracheotomy tube, a drainage tube and six splints. The splints shall be balsa wood 12 inches by 1/2 inch by 4 inches.

3.4.1.3.1.2.7.11 Clothes Storage. - Clothes storage shall provide for 2 changes of clothes per man per mission; allow for storage of soiled clothes with minimum of possible contamination; and allow a maximum storage volume of 1476 cubic inches for clean and dirty clothes.

3.4.1.3.1.2.7.12 Water Management. -

3.4.1.3.1.2.7.12.1 Potable and Filtered Water. - Potable and filtered water for crew usage shall be handled by the water supply subsystem described in paragraph 3.3.3.3.5.

3.4.1.3.1.2.7.13 Waste Management. - Waste management shall allow for the collection and storage of liquid and solid human waste in the following manner.

3.4.1.3.1.2.7.13.1 Solid Waste. - Solid human waste shall be disposed of in the defecation-emesis bags described in paragraph 3.4.1.3.1.2.7.3.

3.4.1.3.1.2.7.13.2 Liquid Waste. - Liquid human waste shall be disposed of through a urine cuff supplied for each crew member. The cuff shall be:

Easy to put on and take off

Comfortable

Able to prevent urine leakage

Easy to clean

3.4.1.3.1.2.7.13.2.1 Relief Valves. - One or three (to be determined) relief tubes shall be supplied. Each tube shall be connected to a vacuum through an appropriate valve so that the urine will be vented overboard without chance of backup.

3.4.1.3.1.2.7.14 Portable Life Support System (NASA Supplied). - A portable life support system (backpack) shall be supplied in conjunction with the pressure suit to provide an environment compatible with life support requirements.

3.4.1.3.1.2.7.15 Clothing (NASA Supplied). - Clothing shall be worn for comfort and protection during the mission. This clothing shall consist of a constant wear garment, a pressure suit and a solar radiation garment.

3.4.1.3.1.2.7.15.1 Constant Wear Garment. - The constant wear garment shall be worn under the pressure suit and in the cabin "shirtsleeve" environment. It shall resemble a summer flying suit and shall be worn by the crewmen at all times. The one-piece garment shall be the primary clothing for the crewmen and shall integrate functions of comfort, minor impacts protection, hygiene, bio-monitoring, and possible electrical heating elements in the event a thermal source is needed for overall pressure suit operation. The constant wear garment shall be free of protruding pockets or attachment hardware and shall include an energy-absorbing fitted hood and headset-microphone communications equipment. Appropriate openings shall be allowed for urination and defecation, with throw-away pads in the apocrine sweat gland regions. The pads shall provide odor and bacteria control without affecting the pressure suit as a whole. The entire undergarment assembly shall be worn under the pressure suit and shall serve as a carrier for communications, heating, bio-monitoring, and other appropriate devices.

3.4.1.3.1.2.7.15.2 <u>Pressure Suit</u>. - The pressure suit shall be worn over the constant wear garment for protection of the crewman. The pressure suit and backpack shall be used for extra-vehicular exploration and maintenance.

3.4.1.3.1.2.7.15.3 <u>Solar Radiation Garment</u>. - The solar radiation garment shall be used as an intermittent wear overgarment for protection against solar radiation as necessary.

3.4.1.3.1.2.7.16 <u>Survival Equipment (NASA Supplied)</u>. - The survival equipment shall be divided into two categories:

(1) <u>Collective Survival Kit</u>: The collective survival kit shall be stored in the upper equipment bay of the Command Module. This kit shall provide necessary survival equipment for the crew while they are together in the Command Module or for post-landing contingencies.

(2) <u>Individual Survival Kit</u>: The individual survival kit shall be stored in the seat pan of the crew couch.

3.4.1.3.1.3 Environmental Control System.- The Command Module Environmental Control System shall control the environment in which the flight crew must operate and provide cooling for items of electronic equipment as required. System operation shall be automatic with a provision for manual control by the flight crew in the event of emergency. The major functional divisions unique to the Command Module are as follows. Additional requirements are presented in paragraph 3.3.3.

3.4.1.3.1.3.1 Atmosphere Conditioning.- Regenerative conditioning of the Command Module atmosphere shall include the removal of debris, carbon dioxide and trace contaminants from the air, addition of sufficient oxygen for metabolic needs, adequate pressure control, and provisions for temperature and relative humidity control. A suitable cooling circuit shall cool critical equipment. Prior to launch of the Spacecraft, cooling functions shall be aided by ground support equipment. During earth return, from the entry interface to 100,000 feet altitude, the Command Module ECS shall perform all necessary functions listed above. From 100,000 feet altitude to mainchute deployment, the ECS shall provide air circulation only. From mainchute deployment to touchdown and during the post-landing phase, the ECS shall provide for cabin ventilation with outside air only.

3.4.1.3.1.3.2 Manual Controls.- The following environmental controls shall be provided:

A manual override for each of the items listed:

- cabin compressors
- suit compressors
- catalytic burner
- humidity controller
- temperature controller

A control shall be provided to activate the entry oxygen supply.

Any other controls that may be determined necessary.

3.4.1.3.1.3.3 Displays.- The following environmental displays shall be provided:

Total Cabin pressure

Cabin temperature

Oxygen partial pressure

Carbon dioxide partial pressure

Carbon monoxide partial pressure

Contaminant trace gas levels

Relative humidity

Glycol flow and temperature

3.4.1.3.1.3.4 Subsystems.- The ECS shall consist of six subsystems as described in paragraph 3.3.3.3.

3.4.1.3.1.4 Launch Escape System.- The normal function of the launch escape system shall be to provide an abort capability throughout countdown, first-stage boost, and for the first few seconds of second-stage firing. After successful ignition of the second-stage booster, the launch escape tower shall be separated and laterally translated from the space vehicle. Redundant functions shall be provided for the tower jettison operation. In the event of an abort mode the launch escape system shall provide the impulse to lift safely and laterally translate the Command Module from the remainder of the space vehicle providing altitude and range for deployment of the earth landing system. The launch escape system shall be jettisoned from the Command Module and propelled away from the Command Module prior to the initiation of earth landing operations. In ascent abort situations below 60,000 feet, the forward heat shield shall be jettisoned with the launch escape tower. Normal tower jettison does not remove the forward heat shield. Operation of the launch escape system shall be dictated by crew response and/or the integrated abort system of the launch vehicle.

The launch escape system shall consist of:

Motors: Pitch

Tower Jettison

Launch Escape

Tower: Frame

Skirt

Flow Separator

Separation System

The design objective weight of the Launch Escape System shall be a maximum of 6500 pounds including ballast. The overall height shall be 400 $\pm$0.5 inches.

3.4.1.3.1.4.1 Launch Escape Motor. - The launch escape motor shall provide the impulse to lift the launch escape vehicle from the remainder of the space vehicle and translate it a safe distance and to an altitude sufficient to allow safe and complete deployment of the earth landing system.

3.4.1.3.1.4.2 Tower Jettison Motor. - The tower jettison motor shall provide the impulse to lift the launch escape system from the Command Module. During an abort mode the tower shall be jettisoned after launch escape motor burnout. During a normal mission the Tower Jettison motor shall remove the LES from the space vehicle at approximately 10 seconds after second stage ignition.

3.4.1.3.1.4.3 Pitch Motor. - The pitch motor shall be mounted in the LES nose cone area and placed normal to the "X" axis to provide a resultant thrust vector in the plus "Z" direction.

3.4.1.3.1.4.4 Crew Operation. - The crew will be responsible for the initiation of the launch escape system operation and the selection of control mode. There will be no responsibility assigned ground control or automatic systems unless there is insufficient time or information for crew action.

3.4.1.3.1.4.5 System Arming. - In the event of launch pad or atmospheric abort, the earth landing system shall be armed by the abort signal. Operation of the landing system may then be automatically controlled by time delays a few seconds after escape system jettison.

3.4.1.3.1.4.6 Electrical Power Requirements. - The electrical power system of the Command Module shall provide all electrical power required by the launch escape system for the duration of all launch pad or atmospheric aborts.

3.4.1.3.1.5 Command Module Reaction Control System. -

3.4.1.3.1.5.1 Function. - The function of the Command Module Reaction Control System shall be to provide three axis control of the Command Module during pre-entry, entry, recovery and abort maneuvers.

3.4.1.3.1.5.1.1 Pre-entry. - The reaction control system shall provide the impulse for roll, pitch and yaw after the Command Module is separated from the Service Module and before the Command Module is subjected to the aerodynamic moments of entry. This function shall rotate the Command Module to the proper angle of attack for atmospheric entry and shall then stabilize the angle of attack during the initial buildup of aerodynamic moment.

3.4.1.3.1.5.1.2 Entry. - The atmosphere entry function shall include the impulse for an attitude control and attitude stabilization function in roll and also an attitude rate damping function in pitch and yaw. A variable flight path shall be obtained by rotating the center of gravity of the Command Module. The path of the Command Module shall be controlled aerodynamically by rolling the lift-force vector about the velocity vector. The pitch and yaw rate damping function shall be used to dampen the oscillations that result from the aerodynamic moment.

3.4.1.3.1.5.1.3 Abort. - The atmospheric abort function shall include the impulse for roll control for the lift orientation of the launch escape system and a pitch and yaw control function that will resist or minimize Command Module tumbling during high altitude abort conditions.

3.4.1.3.1.5.2 System Operation

3.4.1.3.1.5.2.1 General. - The complete Command Module Reaction Control System shall consist of two similar reaction control systems identified as system A and system B. The two similar systems shall be operated simultaneously during normal control operation. In the event of a failure of either system A or system B, the remaining system shall be designed to provide adequate control to safely complete the entry missions. Each individual system shall consist of a pressurized helium storage and distribution system, an oxidizer storage and distribution system, a fuel storage and distribution system, and six rocket engine systems.

3.4.1.3.1.5.2.1.1 Pressurized Helium System. - The pressurized helium system shall be composed of the Category A components listed in Table II and shown schematically in Figure 25. The helium supply shall be contained within one spherical tank which shall be cradle mounted to minimize detrimental tank flexures. During ground service operations and prior to initial system pressurization, the high pressure helium shall be confined to the storage tanks by means of the normally closed squib valve. The squib valve shall contain an integral helium filter to protect the downstream regulators and check valves from harmful foreign particles. The manually controlled normally open solenoid valve shall provide the means of isolating the storage tank from the downstream components in the event of downstream leakage or component failure. Upon command, the squib valve shall open the helium supply to a pressure regulation system consisting of two individual regulators connected in series. The upstream regulator in the series shall reduce the high upstream pressure to a value slightly higher than the required downstream pressure. The second regulator in the series shall reduce the output of the first regulator to the required downstream pressure. If the upstream regulator fails open, the second regulator shall be capable of maintaining the required downstream pressure when subjected to the resulting increase in inlet pressure. Conversely, if the second regulator fails open the outlet pressure of the first regulator shall be within the permissible operating range of the propellant systems. The check valves downstream of the regulators shall prevent the propellants from entering the helium system in the event of failure of a propellant tank positive-expulsion device. The relief valve shall contain an integral helium filter and shall prevent an excessive pressure buildup in the propellant storage tanks. The vent valve shall provide means for venting the helium system downstream of the check valves during propellant servicing operations and helium depressurization operations.

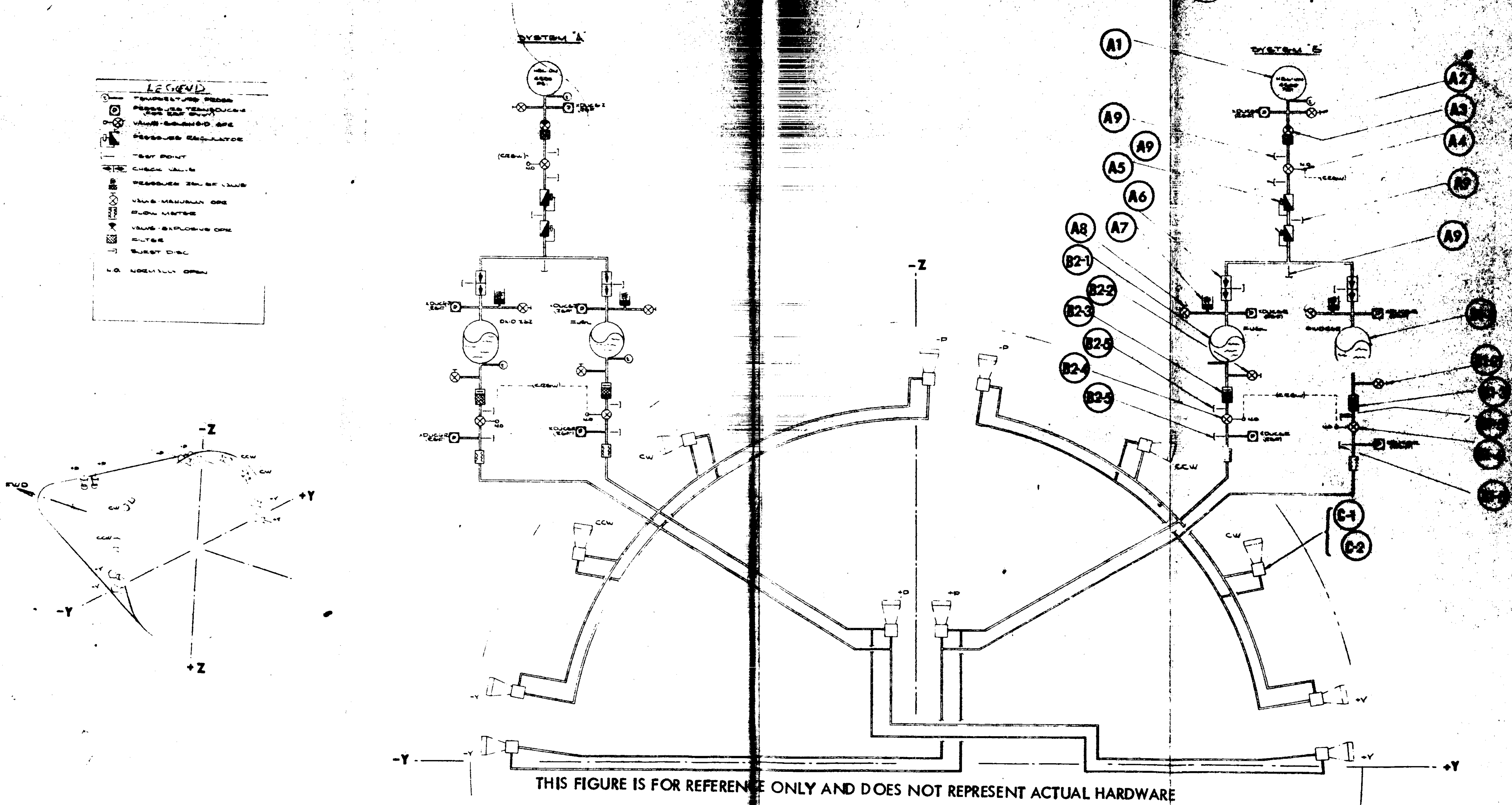

Figure 25. Command Module Reaction Control System

3.4.1.3.1.5.2.1.2 Oxidizer System. - The oxidizer system shall be composed of the category B.1 components listed in Table II and shown schematically in Figure 25. The fill valve shall provide the facility for servicing the oxidizer system during ground operations. The oxidizer supply shall be contained within a hemi-spherically domed cylindrical tank which shall be cradle mounted. The tank shall be equipped with a positive-expulsion device. Pressurized helium from the helium system shall act on the opposite side of the positive-expulsion device forcing the oxidizer through the oxidizer distribution system to the rocket engines at the required feed pressure. During ground service operations and prior to actuation of the helium system squib valve, the oxidizer shall be confined to the storage tank by means of the burst diaphragm assembly. The burst diaphragm assembly shall contain an integral filter to protect the rocket engines from harmful foreign particles. Upon initial pressurization of the system, the pressurized oxidizer shall rupture the burst diaphragm and flow to the normally closed oxidizer injector valves of the rocket engines. The manually controlled, normally open solenoid valve shall provide the means of isolating the oxidizer storage tank from the rocket engines in the event of a leak in the oxidizer distribution system or a malfunction of a rocket engine.

3.4.1.3.1.5.2.1.3 Fuel System. - The fuel system shall be composed of the Category B.2 components listed in Table II and shown schematically in Figure 25. The fill valve will provide the facility for servicing the fuel system during ground operations. The fuel supply shall be contained within a hemispherically domed cylindrical tank which shall be cradle mounted. The tank shall be equipped with a positive-expulsion device.

Pressurized helium shall act on the opposite side of the positive-expulsion device forcing the fuel through the fuel distribution system to the rocket engines at the required feed pressure. During ground servicing operations and prior to actuation of the helium system squib valve, the fuel shall be confined to the storage tank by means of the burst diaphragm to protect the rocket engines from harmful foreign particles. Upon initial pressurization of the system, the pressurized fuel shall rupture the burst diaphragm and flow to the normally closed fuel injection valves of the rocket engines. The manually controlled, normally open solenoid valve shall provide the means of isolating the fuel tank from the rocket engines in the event of a leak in the fuel distribution system or a malfunction of a rocket engine.

3.4.1.3.1.5.2.1.4 Rocket Engine System. - Each rocket engine system shall be composed of the Category C components listed in Table II. Activation of the propellant feed system supplies propellant to the normally closed solenoid inlet valves mounted on the rocket engines. Electrical commands from the stabilization and control system open the oxidizer and fuel valves simultaneously. In the event of failure in the stabilization and control system, the commander commands the engines on a separate manual circuit. The propellants, under system pressure, flow through their corresponding injector passages at high velocity, impinge in the combustion chamber and react exothermically. The expanded high temperature gases flow through the chambers and nozzle system and exhaust through ports faired into the Command Module skin producing the required thrust.

3.4.1.3.1.5.2.1.4.1 Rocket Engine. - The rocket engine shall be a pressure-fed, pulse-modulated, cold wall thrust generator and have the following characteristics:

Thrust. - During continuous operation the rocket engine shall develop a vacuum thrust of 100 plus or minus 5 pounds.

Thrust Transient Rate. - The rocket engine shall demonstrate a thrust buildup and decay as shown in Figure 26.

Specific Impulse. - The rocket engine shall achieve the following specific impulses during operation under vacuum conditions.

Continuous Operation. - The rocket engine shall develop a specific impulse of at least 300 seconds when operating for periods in excess of one second.

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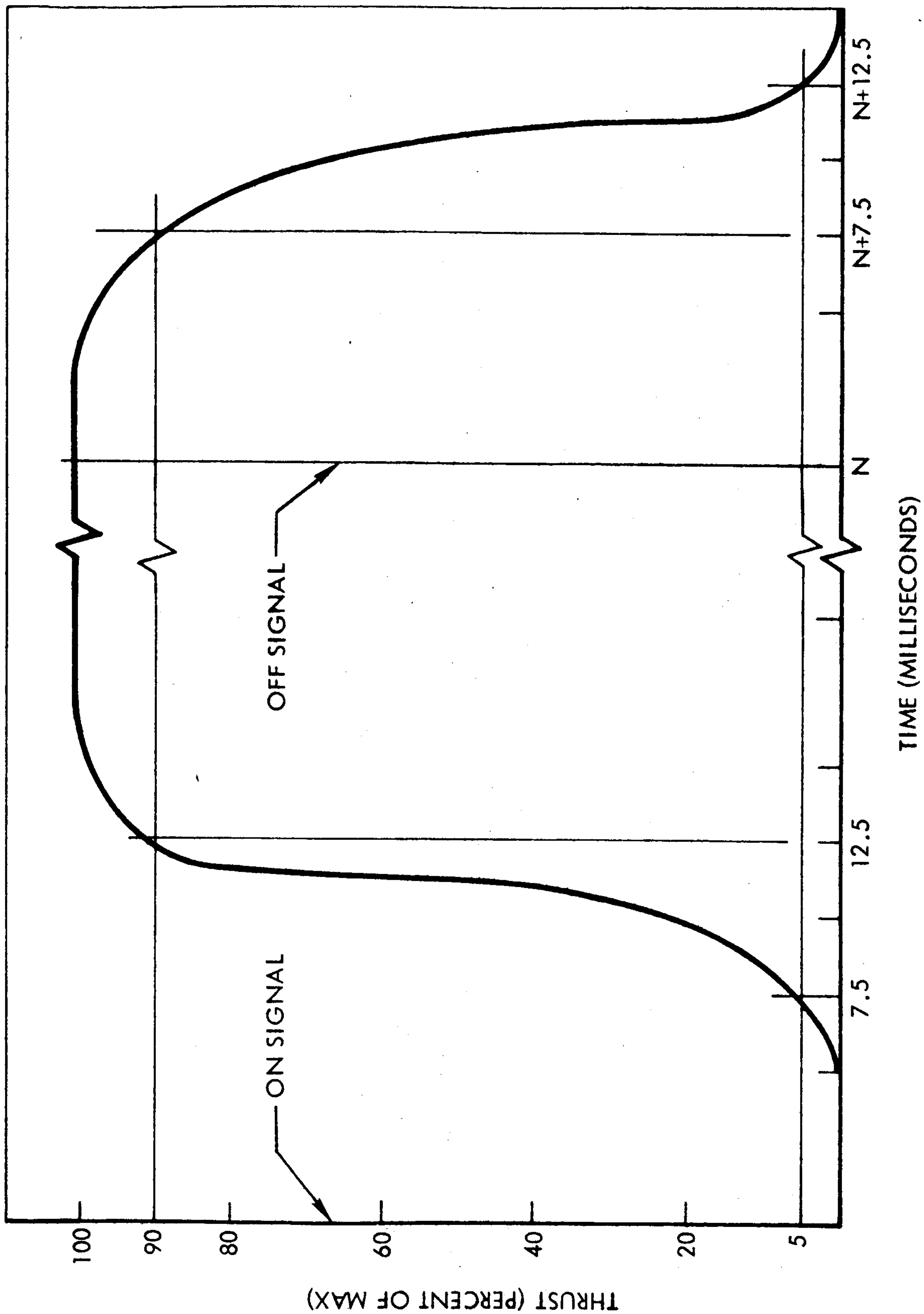

Figure 26

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Pulse Mode Operation. - The rocket engine shall develop the specific impulse levels shown in Figure 27 when operating at pulse widths less than one second.

3.4.1.3.1.5.3 Component Performance. - A summary of the functional requirements for each category A, B and C component is outlined in Table II. Performance requirements which apply to each component in general, are as follows:

3.4.1.3.1.5.3.1 Fluid Compatibility. -

3.4.1.3.1.5.3.1.1 Category A Components. - All Category A (pressurization system) components shall be compatible with high-grade oil-free commercial helium for long periods of exposure and intermittent exposures of short duration.

3.4.1.3.1.5.3.1.2 Category B.1 Components. - All Category B.1 (oxidizer system) components shall be compatible with nitrogen tetroxide (N_2O_4) for long periods of exposure and intermittent exposures of short duration.

3.4.1.3.1.5.3.1.3 Category B.2 Components. - All Category B.2 (fuel system) components shall be compatible with a mixture of 50 percent hydrazine (N_2H_4) and 50 percent unsymmetrical dimethylhydrazine (UDMH) for long periods of exposure and intermittent exposures of short duration.

3.4.1.3.1.5.3.1.4 Category C Components. - All Category C (engine system) components shall be compatible with the fluids specified for Categories B.1 and B.2.

3.4.1.3.1.5.3.2 Pressures. - Helium source pressure for the pressurization system shall be approximately 4500 psig. Propellant shall be distributed to the engine at an approximate operating pressure of 170 psia.

3.4.1.3.1.5.3.3 Leakage. - All components shall be designed to perform as required with zero leakage.

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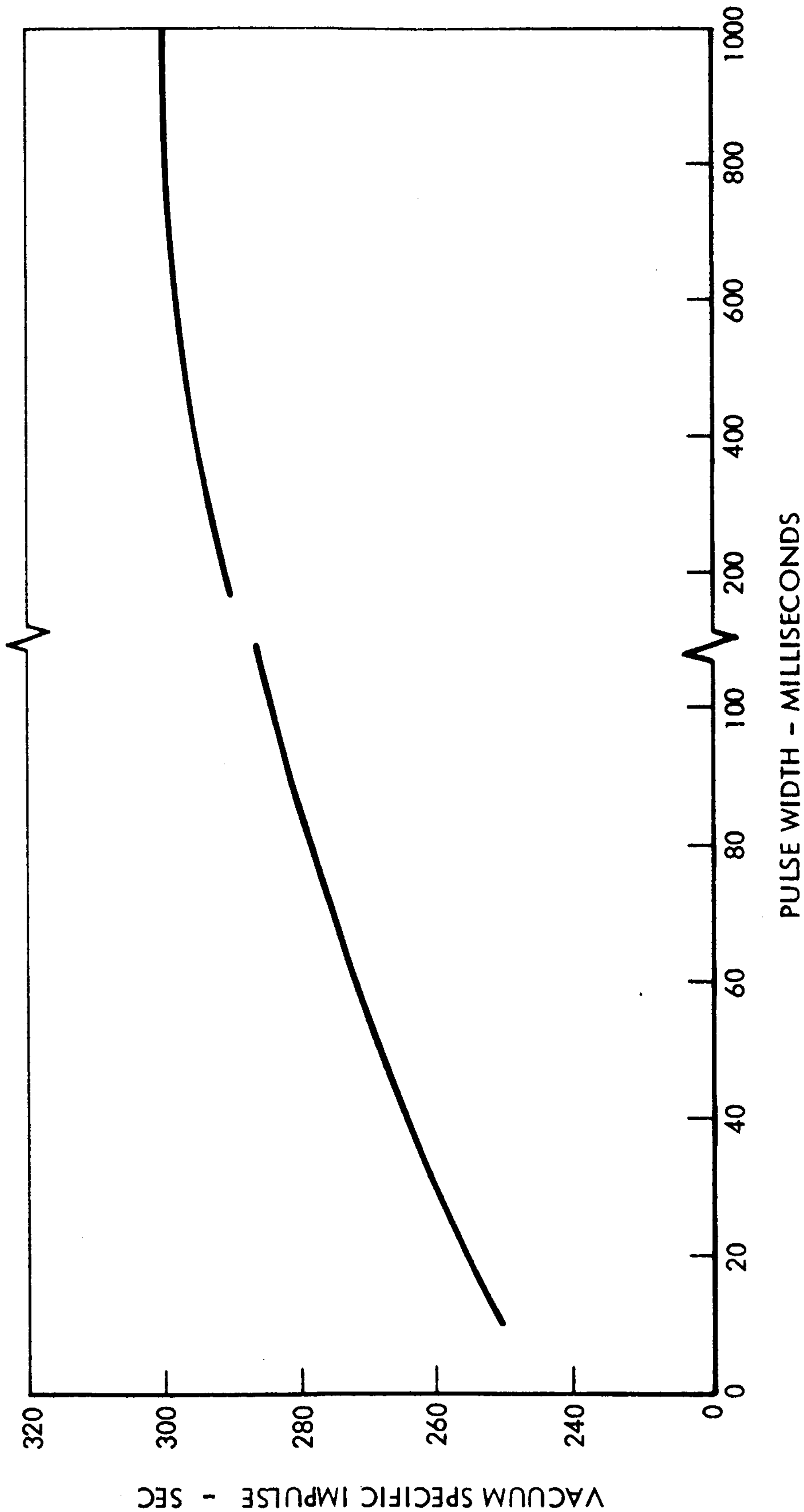

Figure 27

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Table II. Command Reaction Control Subsystem Components

Category	Figure 25 Code No.	Component Title	Function
A	A-1	Vessel - Helium Pressure	Storage of high-pressure helium
A	A-2	Valve - Helium Fill, Manual	Fill point during ground servicing operations
A	A-3	Valve - Helium, Squib	Confine high-pressure helium to storage area during ground servicing operations
A	A-4	Valve - Helium, Solenoid Operated	Isolate the storage area in the event of a down-stream failure
A	A-5	Regulator - Helium Pressure	Maintain the required constant downstream pressure
A	A-6	Check Valve - Helium Pressure	Prevent oxidizer and/or fuel from backing up into helium system
A	A-7	Relief Valve - Helium Pressure	Prevent over-pressurization of fuel and oxidizer system
A	A-8	Valve - Vent, Helium Pressure	Depressurize low pressure side of helium system
A	A-9	Coupling - Check, Helium Pressure	Provide pressure check point

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Table II. Command Reaction Control Subsystem Components (Cont.)

Category	Figure 25 Code No.	Component Title	Function
B.1	B.1-1	Tank - Oxidizer	Storage of nitrogen tetroxide (N_2O_4)
B.1	B.1-2	Valve - Oxidizer Fill, Manual	Fill point during ground servicing operation
B.1	B.1-3	Burst Diaphragm - Oxidizer	Confine N_2O_4 to storage area until it is pressurized
B.1	B.1-4	Valve - Oxidizer, Solenoid Operated	Isolate storage area in the event of downstream failure
B.1	B.1-5	Coupling - Check Oxidizer	Provide pressure check point
B.2	B.2-1	Tank - Fuel	Storage of a 50/50 blend of UDMH and hydrazine (N_2H_4)
B.2	B.2-2	Valve - Fuel Fill, Manual	Fill point during ground servicing operation
B.2	B.2-3	Burst Diaphragm - Fuel	Confine fuel to storage area until it is pressurized
B.2	B.2-4	Valve - Fuel, Solenoid Operated	Isolate storage area in the event of downstream failure
B.2	B.2-5	Coupling - Check, Fuel	Provide pressure check point

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Table II. Command Reaction Control Subsystem Components (Cont.)

Category	Figure 25 Code No.	Component Title	Function
C	C-1	Propellant Valves (2)	The valves are electrically actuated solenoid valves matched to provide simultaneous propellant injection to the thrust chamber
C	C-2	Thrust Chamber	The thrust chamber combines the hypergolic propellants resulting in an exothermic reaction. The resulting high pressure gases will be ejected through a nozzle and the gas momentum will be converted into thrust.

3.4.1.3.1.5.3.4 <u>Power Supply</u>. - All components which are electrically actuated shall operate from a power supply having the following characteristics:

Steady State Voltage	25-30 volts dc
Transient Voltage Limits	25-30 volts dc. Recovery time from one steady-state level to another upon load changes is less than 0.7 sec.
Ripple Voltage	250 millivolts peak-to-peak maximum

3.4.1.3.1.5.3.4.1 <u>Grounding</u>. - All components which are electrically actuated shall not be internally grounded.

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3.4.1.3.1.5.3.4.2 Dielectric Strength. - All components which are electrically actuated, shall withstand 1500 volts (RMS) at commercial frequency, for a period of one minute (at sea level), without evidence of insulation breakdown or flashover.

3.4.1.3.1.5.3.4.3 Insulation Resistance. - The insulation resistance for each components, which is electrically actuated, shall be a minimum of 100 megohms, when a potential of 500 volts dc is applied for a period of two minutes (at sea level).

3.4.1.3.1.5.4 Attitude and Stabilization. - The Spacecraft reaction control system shall accept manual and automatic attitude control and attitude stabilization signals from both the guidance and navigation system and the stabilization and control system.

3.4.1.3.1.5.4.1 Electrical Power Requirements. - The Command Module and Service Module electrical power system shall provide power to the reaction control system at all times except when the Command Module is separated from the Service Module.

3.4.1.3.1.5.4.2 Touchdown Positioning. - Prior to touchdown, the Command Module shall be positioned by the Command Module reaction control system to produce the most favorable Command Module attitude for impact attenuation.

3.4.1.3.1.6 Guidance and Navigation (G&N) System (NASA Supplied). - The G&N system shall be designed and manufactured by associate contractors designated by NASA. NAA shall be responsible for the interface design and for integrating the G&N system into the Spacecraft. NAA shall specify the overall reliability, environmental and performance requirements of the G&N system.

3.4.1.3.1.6.1 Functions. - The major functions of the Command Module G&N system are: (1) a primary intertial reference; (2) acceleration, velocity and position determination; (3) G&N computation and prediction; (4) Provide attitude and thrust inputs to the stabilization and control system for primary and abort modes. The primary attitude reference will be established before lift-off and re-established by electronic and/or optical sighting means during the flight phases. The acceleration, velocity and position function shall include the measurement and computation of Command Module and Service Module acceleration, velocity and position data so that these data may be used to manually or automatically control velocity and course changes required to meet navigational and steering requirements of the APOLLO missions.

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The navigation computation and prediction function shall include the compilation and monitoring of the moving position of the Command Module and Service Module with respect to a number of coordinate axes including those of the earth and moon. The system shall also be capable of calculating and displaying steering and velocity changes that are required to accomplish the next segment of a particular APOLLO trajectory. The function of providing attitude and thrust inputs to the S&C system shall include attitude stabilization during navigation sightings, and attitude commands for powered phases of the mission. In addition, this function will provide thrust initiation and cutoff signals during thrusting. At all times during the extra-atmospheric mission phases the G&N system shall be capable of generating attitude and thrust commands to accomplish an abort.

3.4.1.3.1.6.2 Inertial Reference. - The G&N system shall be able to provide its own primary inertial reference and be compatible with a secondary inertial reference within the stabilization and control system. It shall provide attitude and thrust control signals to the stabilization and control system. The Guidance and Navigation system shall be capable of detecting and monitoring any attitude and velocity vector changes of the Command Module and Service Module initiated as a result of the execution command given. The navigational data needed to accomplish an abort during any phase of the APOLLO mission shall be generated and supplied by the G&N system. The indication for abort, the point of abort initiation, and associated data necessary for abort shall be displayed to the flight crew.

3.4.1.3.1.6.3 Guidance Phases. - The Guidance and Navigation system shall be capable of providing the necessary functions required to guide and navigate an APOLLO Command Module and Service Module throughout an APOLLO mission except during boost. The guidance during the earth ascent phase injection into the earth parking orbit and translunar injection phases shall be provided by the booster guidance system. The G&N system shall be capable of monitoring the entire earth launch operation and displaying attitude and trajectory information. Upon completion of the translunar injection phase, the G&N system shall assume the C/M and S/M guidance and provide primary guidance and navigation throughout the remainder of the mission independent of the GOSS communications.

3.4.1.3.1.6.4 In-Flight Tests. - In-flight tests and maintenance capability shall be provided within the Guidance and Navigation system to allow monitoring and maintenance of the Guidance and Navigation system. Displays and controls shall be provided to form the interface between the crew and the Guidance and Navigation system.

3.4.1.3.1.6.5 Optical System. - The optical system shall aid in performing the Guidance and Navigation functions. It shall consist of a sextant and a scanning telescope.

3.4.1.3.1.6.5.1 Sextant. - The sextant shall be a high power, narrow field of view guidance and navigation device which measures the angle between two targets. It shall be slaved to the scanning telescope and physically aligned to the inertial measurement unit. It shall be located on the forward sidewall of the lower equipment bay.

3.4.1.3.1.6.5.2 Scanning Telescope. - The scanning telescope is a low power, wide angle device used to acquire and position stars and landmarks within the narrower field of voew of the sextant. It shall be located adjacent to, and aligned to the sextant.

3.4.1.3.1.7 Earth Landing System. - The earth landing system shall utilize a parachute technique for landing the Spacecraft.

3.4.1.3.1.7.1 Function. - The earth landing system shall provide Command Module stabilization and reduce vertical velocity to allow a slow and safe touchdown of the Command Module from any mission or abort operation as shown in Figure 28.

3.4.1.3.1.7.2 Parachute System. - The parachute system shall be a logical and orderly extension of the present state-of-the-art of landing systems. Multiparachute capability shall provide redundancy for the landing system. The system shall have the following function capabilities.

3.4.1.3.1.7.3 Spacecraft Stabilization. - The parachute system shall stabilize the Command Module during the post-entry phase. Stabilization shall be accomplished by a drogue parachute during early descent and by landing parachute during the remainder of descent. The drogue parachute shall provide stabilization with a Command Module tumbling rate of up to 50^{o}/second.

3.4.1.3.1.7.4 Velocity Control. - The landing parachutes shall reduce the vertical velocity of the Command Module to not more than 34 ft/sec at touchdown with two of the parachutes operating.

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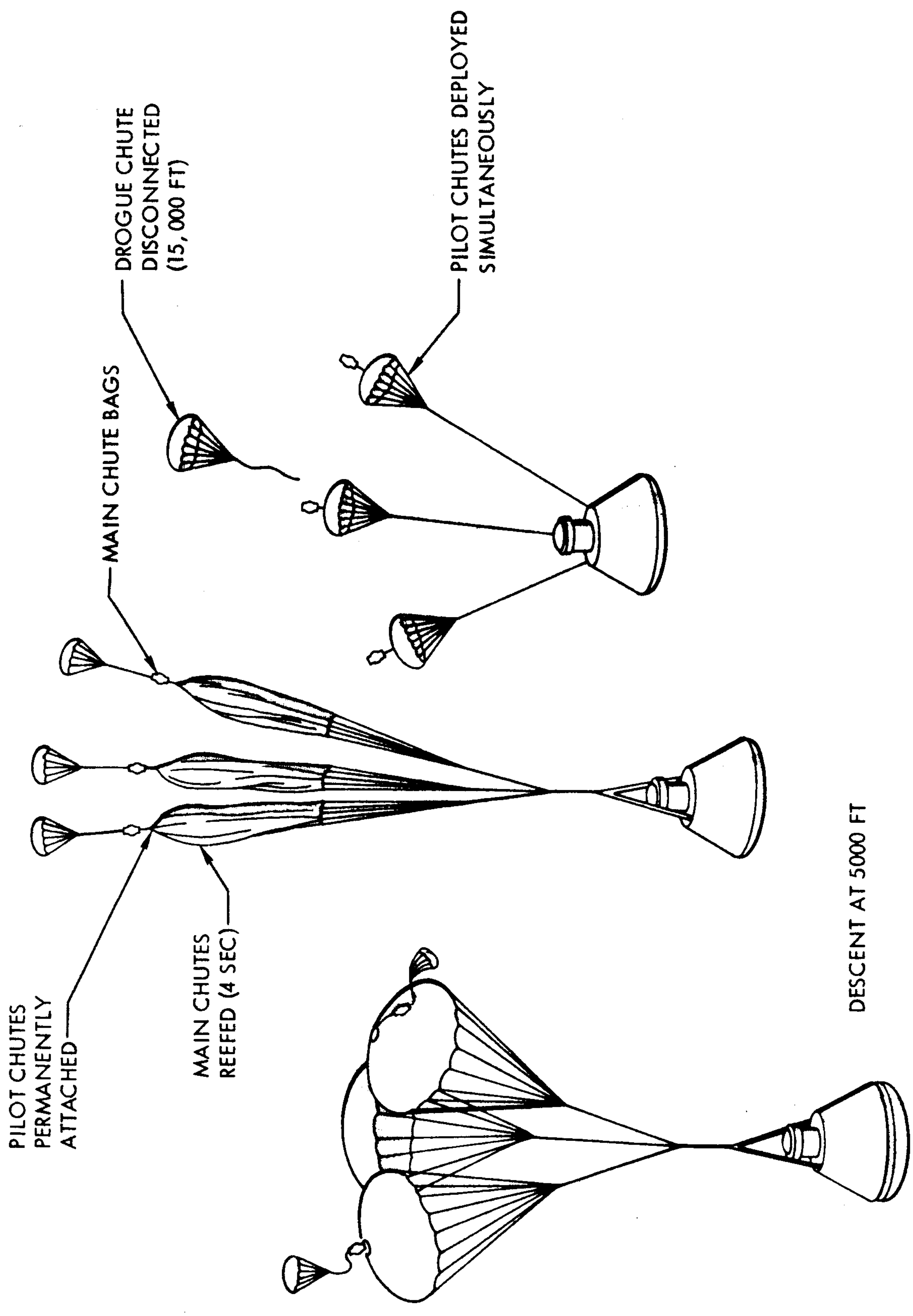

Figure 28. Main Parachute System Deployment Sequence

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3.4.1.3.1.7.5 Impact Attenuation. - The Impact Attenuation system shall reduce touchdown shock such that the Command Module primary structure or flotation is not impaired. Any further attenuation required by the crew shall be provided by individual crewman shock-attenuation devices. (Reference Figure 29.)

3.4.1.3.1.7.6 Recovery Aids. - The Recovery Aids shall provide location and survival aids necessary for safe and prompt recovery of the Command Module and crew.

3.4.1.3.1.7.7 System Arming. - During normal entry, the earth landing system shall be armed by the crew at the initiation of the entry phase. In the event of launch pad or atmospheric abort, the earth landing system shall be armed by an abort signal from the launch escape system. Operation of the landing system during abort shall be controlled by time delays a few seconds after escape tower jettison.

3.4.1.3.1.7.8 Crew Operation. - The crew shall have the prerogative of manually or automatically controlling the initiation of all functions.

3.4.1.3.1.7.9 Impact Attitude Control. - Prior to touchdown, the Command Module shall be positioned by the Command Module reaction control system to produce the most favorable Command Module attitude for impact attenuation.

3.4.1.3.1.7.10 Electrical Power Requirements. - The Command Module electrical power system shall provide electrical power to the earth landing system which activates the selected sequence of operations of the various earth-landing subsystems.

3.4.1.3.1.8 Command Module Electrical Power System. -

3.4.1.3.1.8.1 Entry Battery System. - Two zinc silver oxide batteries shall be provided to supply all Command Module electrical power requirements during entry and recovery flight phases. The capacity of each entry battery shall be sufficient to supply essential loads during entry and recovery in the event of failure of the other entry battery. Entry batteries shall also be used to supply peak system load demands during earlier flight phases. At all other times the entry batteries shall be isolated from load buses. One battery charger shall be provided to recharge the entry batteries from fuel cell subsystem power after such use.

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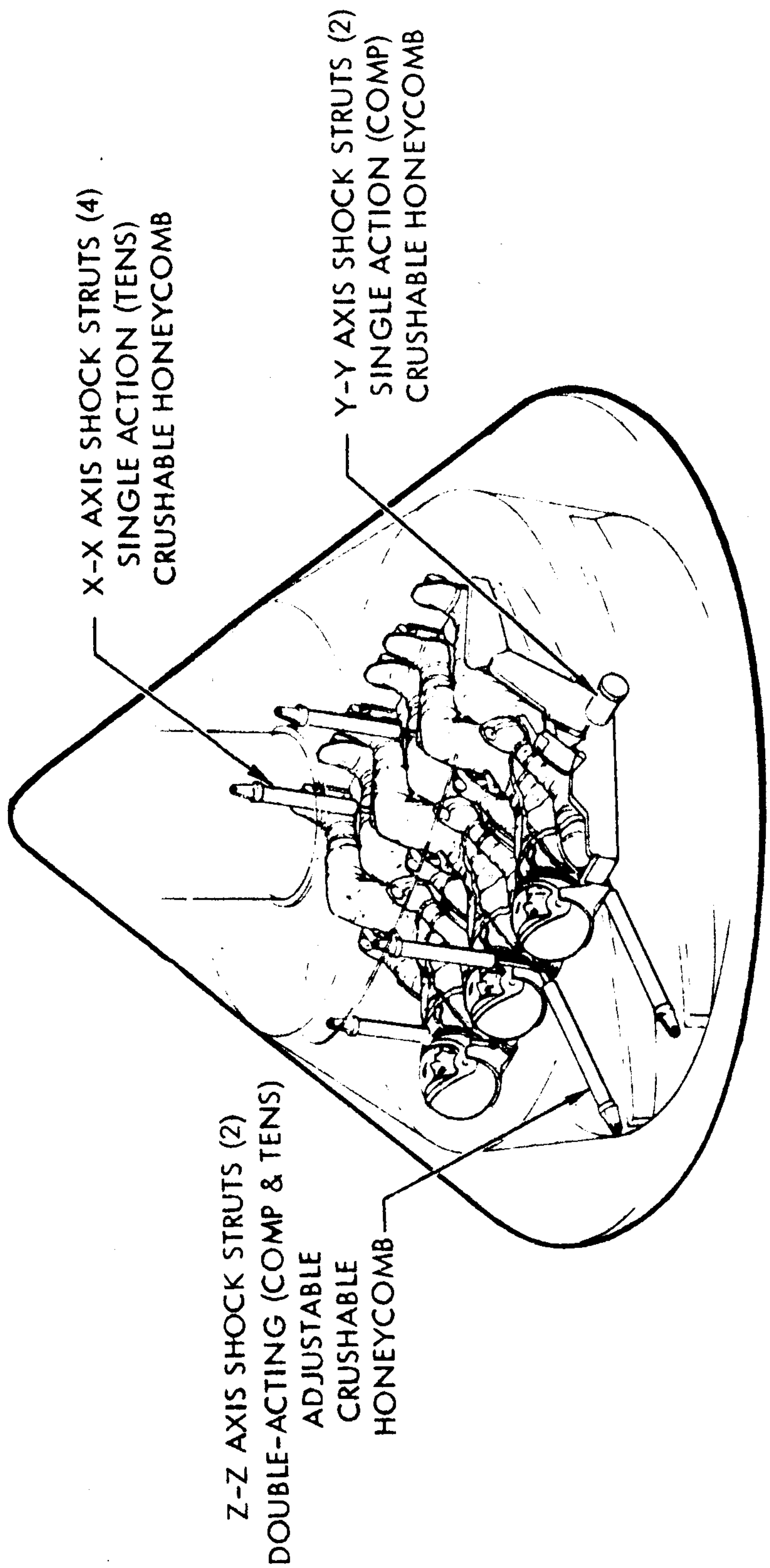

Figure 29. Impact Attenuation Crew System

3.4.1.3.1.8.2 Post Landing Battery System. - One zinc silver-oxide battery shall be provided to supply essential loads during the post-landing phase. This battery shall be fully charged normally and isolated from the main electrical power system. Provisions shall be made to recharge the post-landing battery in flight from the entry battery charger for emergency conditions such as excessive battery temperature. For normal post-landing conditions the unused capacity of the entry batteries shall be made available for post-landing loads.

3.4.1.3.1.8.3 Electrical Power Distribution. -

3.4.1.3.1.8.3.1 Load Classification and Power Source Operating Modes. - All electrical loads supplied by the distribution shall be classified as Essential, Non-Essential, Pyrotechnic, or Post Landing. Essential loads are defined as those loads, (except pyrotechnic circuits) which are mandatory for safe return of the Command Module and Service Module from any point in any mission. Non-Essential loads are defined as those which are necessary to the successful completion of the mission. Pyrotechnic loads are Essential loads employing pyrotechnic devices. Post landing loads are those required after earth landing.

3.4.1.3.1.8.3.2 Distribution - DC Power. - The output of all three fuel cell modules and both entry batteries shall be fed to each of two isolated redundant Essential buses, A and B as shown in Figure 4. Redundant Essential loads shall be connected alternately to Essential bus A or B as shown. Non-redundant Essential loads shall be connected to both Essential buses, with isolation diodes to retain isolation of Essential buses A and B. Redundant pyrotechnic buses shall be provided and supply only pyrotechnic loads. All Essential loads shall be so connected to the two Essential buses that loss of power to either bus will not cause interruption of power to any Essential load system. The Non-Essential loads shall be connected to the Non-Essential bus and provision made for manually disconnecting these loads as a group under emergency conditions. The Non-Essential bus shall be connected to both Essential buses through isolating diodes. Post-landing loads shall be supplied by the post-landing bus.

3.4.1.3.1.8.3.4 Inversion and Distribution - AC Power. - Three static inverters shall be provided to supply 400 cps, 115/200 volts, 3 phase ac power by inversion from the spacecraft primary dc power sources. Only one inverter shall operate and have capacity for supplying all Spacecraft primary 400 cps power, with the other two inverters acting as redundant idle standby units. Transfer relays shall manually switch input and loads to one of the other inverters in event of inverter failure. Inverters shall supply ac buses No. 1 and No. 2 as shown in Figure 4, to which alternate Essential redundant ac loads shall be connected. Provision shall be made to disconnect and isolate one of these buses from the inverter and the other bus in the event of bus failure.

3.4.1.3.1.8.3.5 Power Returns and System Grounding. - Power distribution from the power sources shall be by means of a two-wire system for dc loads and single phase ac loads, and four-wire system for three-phase ac loads. The power return current shall be through the second dc wire (or single phase ac load wire) or the fourth ac wire for unbalanced three-phase ac loads, to a negative dc bus and a neutral ac bus. These two buses shall have the only ground connection to Spacecraft structure in the Spacecraft primary ac and dc power system. This ground connection shall not be interrupted by any control or switching devices. The negative and neutral return connections to primary power in each load system shall be isolated from structure ground and from each other.

3.4.1.3.1.8.3.6 Circuit Protection. - Circuit protection for the primary power system shall be provided by the system of circuit breakers and diodes. Circuit breakers shall be provided for each feeder off of every bus for the following purposes:

(a) To clear the buses of ground faults in the load system feeder wiring, or other buses.

(b) To protect feeder wiring from deterioration (including production of smoke or toxic fumes in the command module) caused by faults or overloads.

3.4.1.3.1.8.3.7 Power Distribution System Wiring, Control and Display. - The power distribution system shall include all necessary connectors, wiring, switching, relays, controls, and indicating devices for controlling electrical power sources, indicating status thereof, supplying and controlling power to all electrical systems, and interconnecting wiring between modules of electrical systems.

The distribution panels shall be maintained at ground potential, (Dead front) and adequately enclosed or otherwise protected to minimize hazards to the crew and provide maximum protection for the electrical system and components. Switching and control shall be accomplished by manually operated circuit breakers or contactors in preference to electrically operated contactors, except where the use of a remotely controlled device is necessary to reduce conductor weight or locate circuit protection or control at the appropriate place.

3.4.1.3.1.9 Communication and Instrumentation. - Refer to paragraph 3.3.1.

3.4.1.4 Interfaces. - Refer to paragraph 3.5

3.4.1.5 Environments. - Refer to paragraph 3.10

3.4.2 Service Module. -

3.4.2.1 Utilization. - Service Module shall provide propulsion for the Command Module and Service Module partial Spacecraft.

3.4.2.2 Dimensions. -

3.4.2.2.1 Geometric Parameters. -

Overall height	= 155 $\pm$0.5 inches
Maximum diameter	= 154 $\pm$0.5 inches
Structure Outline	= See Figure 30

3.4.2.2.2 Weight. - The following weights represent design objectives:

Gross Burnout	= 11,000 lbs.
Usable Propellant weight	= 45,000 lbs. maximum

3.4.2.3 Area Designations. - See Figure 30.

3.4.2.4 Performance. - The Service Module shall be designed to provide propulsion increments during various mission phases.

3.4.2.4.1 Requirements. - The Service Module shall be capable of operating in the space environments defined in paragraph 3.10. The Service Module performance requirements are defined under the respective systems paragraphs.

3.4.2.5 Systems. - The Service Module consists of a:

(a) Service Propulsion System

(b) Reaction Control System

(c) Communications and Instrumentation System

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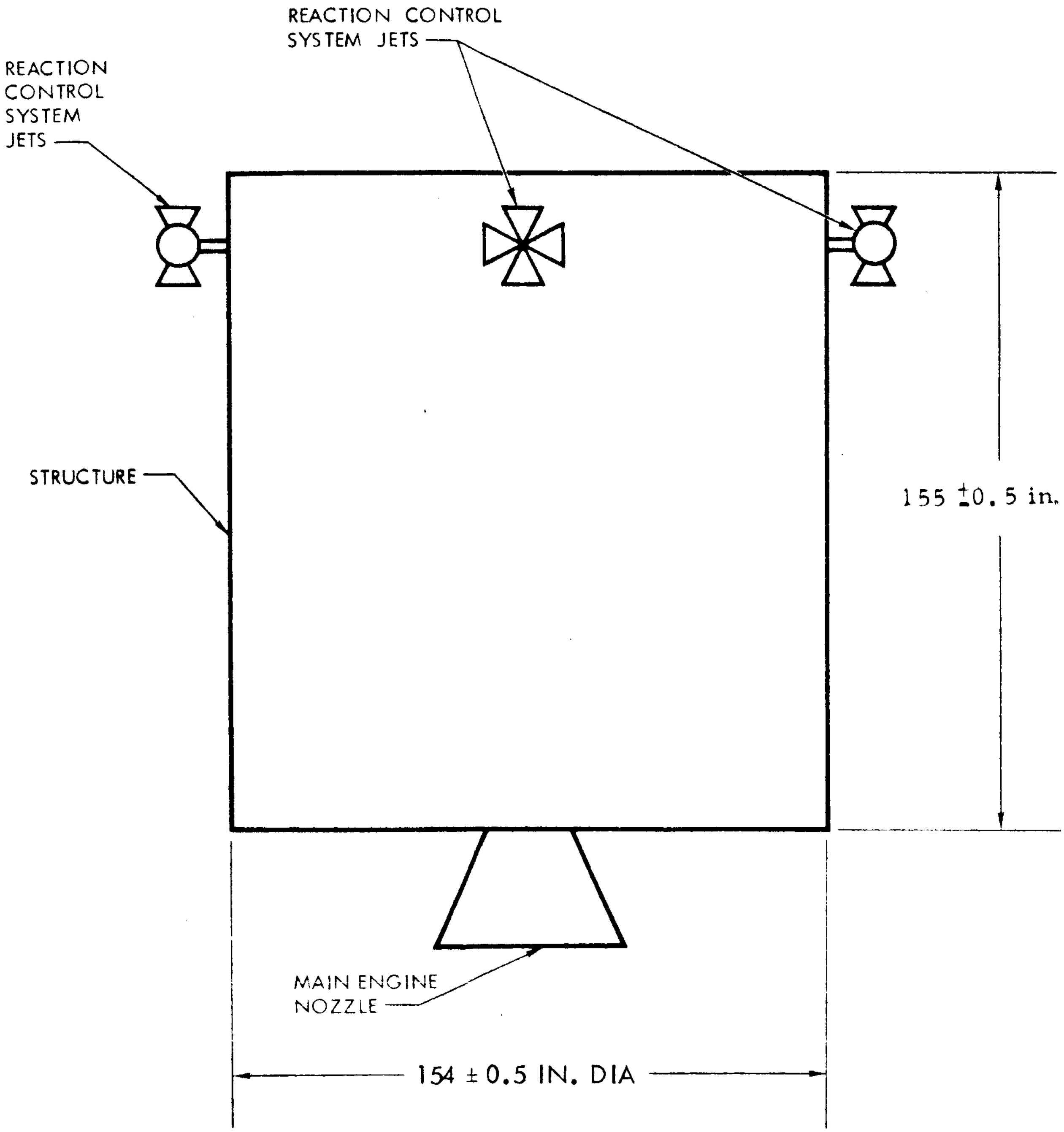

Figure 30. Service Module

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(d) Structural System

(e) Environmental Control System

(f) Electrical Power System

3.4.2.5.1 Service Propulsion System. - The Service Propulsion System shall function in several normal and emergency modes.

3.4.2.5.1.1 General. - The service propulsion system shall consist of a helium pressurization system, a hypergolic propellant system (N_2O_4/Aerozine) and an engine system.

3.4.2.5.1.1.1 Pressurization System. - The pressurization system shall be composed of the Category A components listed in Table III and shown schematically in Figure 31. Helium gas shall be utilized for the pressurization system fluid medium. The helium supply shall be contained within two spherical tanks, cradle-mounted for the purpose of eliminating the possiblity of detrimental tank skin flexures. Two pressure regulation systems, installed in parallel, shall be located downstream of the helium storage tanks. Each of these systems shall contain a primary and secondary pressure regulator, located in series. The upstream primary regulator in the series shall reduce the high upstream pressure to a value slightly higher than the required downstream pressure. The secondary downstream regulator in the series shall reduce the output of the primary regulator to the required downstream pressure. If the primary regulator fails open, the secondary regulator shall be capable of maintaining the required downstream pressure when subjected to the resulting increase in inlet pressure. Conversely, if the secondary regulator fails open, the outlet pressure of the primary regulator shall be within the permissible operating range of the propellant systems. A manually controlled, normally open solenoid valve shall be installed upstream of each pressure regulation system. The actuation of these valves shall provide the means of isolating the helium tank in the event of downstream leakage or component failure. Helium from the pressure regulation system shall enter two check valves, located in parallel and upstream from the main oxidizer and propellant tanks, respectively. The incorporation of these valves shall prevent the contact fuel and oxidizer after initial pressurization of the system. For redundancy, each check valve assembly shall consist of four integral check valves. One pressure relief valve and one burst diaphragm-integral filter component shall be provided

TABLE III - SERVICE PROPULSION SYSTEM COMPONENTS

Category	Figure Code No.	Component Title	Component Function
A			
	A-1	Vessel-Helium Pressure	Storage tank for high pressure helium gas
	A-2	Disconnect-Helium Fill and Drain, Manual	Helium fill and drain point during ground servicing operations
	A-3	Coupling-Helium Pressure, Check	Provides pressure check point during checkout operations
	A-4	Valve-Solenoid Operated, N.O. Helium Pressure	Isolates the helium storage tank in the event of downstream leakage or component failure
	A-5	Regulator-Primary, Helium Pressure	Maintains the required downstream pressure
	A-6	Regulator-Secondary, Helium Pressure	Maintains the required downstream pressure
	A-7	Valve-Helium Pressure Check	Prevents the contact of fuel and oxidizer
	A-8	Valve-Helium Pressure Relief	Maintains structural integrity of propellant tanks in the event of pressure rise in tankage system
	A-9	Diaphragm-Helium Pressure Burst	Prevents propellant from entering relief valve, prior to rupture
B			
B. 1	B.1-1	Disconnect-Oxidizer Vent, Manual	Provides venting of propellant tanks during ground servicing operations
	B.1-2	Tank-Oxidizer	Storage tank for N_2O_4 oxidizer supply
	B.1-3	Disconnect-Oxidizer Fill and Drain, Manual	Oxidizer fill and drain point during ground servicing operations Positively seals oxidizer supply in storage tanks prior to initial system pressurization
	B.1-4	System-Propellant Utilization	Controls oxidizer flow to maintain 2:1 oxidizer to fuel ratio
B. 2	B.2-1	Disconnect-Fuel Vent, Manual	Provides venting of propellant tanks during ground servicing operations
	B.2-2	Tank-Fuel	Storage tank for 50% N_2H_4 and 50% UDHM fuel supply
	B.2-3	Disconnect-Fuel Fill and Drain, Manual	Fuel fill and drain point during ground servicing operations Positively seals fuel supply in storage tanks prior to initial system pressurization
C			
	C-1	Thrust Chamber Assembly	Combustion chamber
	C-2	Valve Assembly-Fuel	Serves as amin propellant valve to control fuel flow to the combustion chamber
	C-3	Valve Assembly-Oxidizer	Serves as main propellant valve to control oxidizer flow to the combustion chamber
	C-4	Actuator Assembly-Hydraulic	Controls position of gates in fuel and oxidizer main propellant valves
	C-5	Valve Assembly-Solenoid	Controls fuel flow to the hydraulic actuators
	C-6	Valve Assembly-Check	Assures maintenance of pressure in actuator overboard feeder lines
	C-7	Disconnect Assembly	Servicing point for applying fuel pressure for ground checkout of engine valving

THIS FIGURE IS FOR REFERENCE ONLY AND DOES NOT REPRESENT ACTUAL HARDWARE

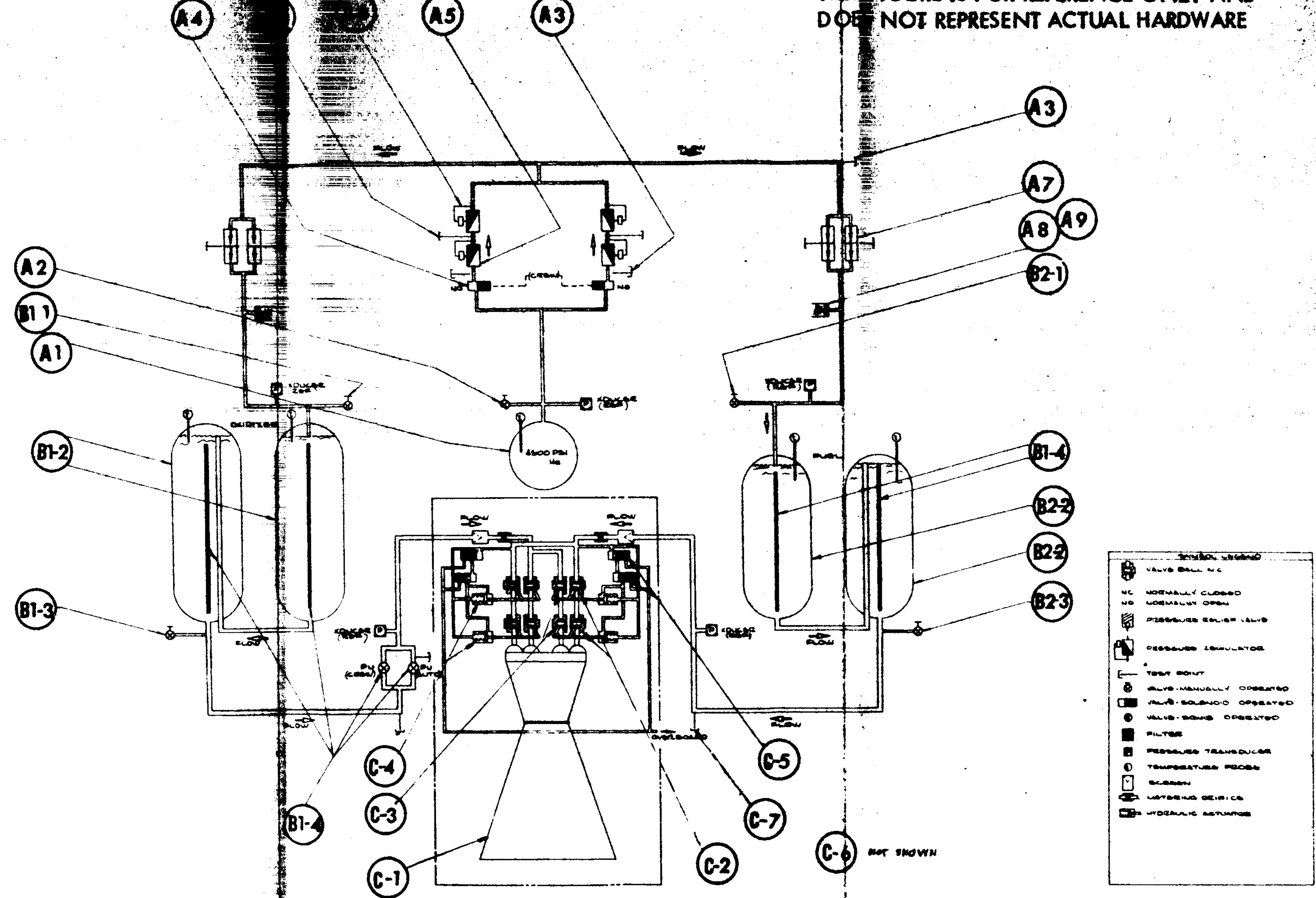

Figure 31. Service Propulsion System

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for each of the oxidizer and fuel dual tankage systems described below. The two pressure relief valves shall maintain the structural integrity of the propellant tanks in the event of an excessive pressure rise in the dual tankage systems. Prior to being ruptured by excessive tank pressure, the burst diaphragms shall prevent propellants from entering the relief valves. The integral filters shall prevent foreign particles from entering the relief valves after rupture of the burst diaphragms occurs.

3.4.2.5.1.1.2 Propellant System. - The propellant system shall consist of the oxidizer (N_2O_4) and fuel (50/50 UDMH and N_2H_4) supply systems. Both systems, except as noted below, shall be functionally and physically similar.

3.4.2.5.1.1.2.1 Oxidizer System. - The oxidizer system shall be composed of the Category B.1 components listed in Table III and shown schematically in Figure 31. The oxidizer supply shall be contained within two hemispherically domed, cylindrical tanks which shall be structurally mounted and connected in series. Each of the tanks shall contain oxidizer liquid level sensing devices. Positive expulsion devices shall not be utilized in the system. During a zero "g" condition, the initial ullage acceleration supplied by the service module reaction control system, shall force the oxidizer to the bottom of the tanks. The oxidizer supply to the engines shall therefore be provided by gravity feed methods. A propellant utilization system, which will control oxidizer flow, shall be provided to assure simultaneous oxidizer and fuel depletion. Signals to the system shall be provided by the oxidizer (and fuel) liquid level sensing devices. A single oxidizer distribution line from the dual tankage system to the propellant utilization system shall be provided.

3.4.2.5.1.1.2.2 Fuel System. - The fuel system shall be composed of the Category B.2 components listed in Table III and shown schematically in Figure 31. The components comprising the fuel system shall be functionally and physically similar to those components comprising the oxidizer system with the following exceptions:

(a) The two fuel tanks shall contain approximately half the volume of the oxidizer tanks due to the required propellant supply ratios.

(b) A propellant utilization system shall not be incorporated within the fuel system. Fuel shall flow from the tanks to the engine in a similar manner as described for the oxidizer system, except said flow shall be direct.

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3.4.2.5.1.1.3 Engine System. -

3.4.2.5.1.1.3.1 General. - The service propulsion system engine shall be a liquid fueled rocket engine employing earth storeable hypergolic propellants. The engine shall be pressure-fed, non-throttleable, and shall contain gimbal provisions for thrust vector control. Chamber cooling shall be accomplished by the use of an ablative combustion chamber fabricated of phenolic refrasil. A radiation cooled nozzle extension shall be utilized to increase propulsion efficiency.

3.4.2.5.1.1.3.2 System Description. - Propellant flow from the propellant supply system shall be divided at the inlet to the engine into parallel fuel and parallel oxidizer lines. Series mounted ball-type shutoff valves shall be installed in each fuel and oxidizer line. Each set of valves (one oxidizer, one fuel) in the parallel paths shall be mechanically interconnected to assure synchronized operation. The mechanically interconnected valve set shall be operated by a single hydraulic actuator utilizing fuel as the actuating medium. Propellant flow to the hydraulic actuators of each redundant valve loop is controlled by two normally closed solenoid valves mounted in series. The valving system, as shown schematically in Figure 31, shall employ redundancy as required to assure engine shutdown in event of a single valve failure in the open position and to allow an engine start in event of a single failure in the closed position.

3.4.2.5.1.1.3.3 Operation. - Engine start shall be signaled by electrical command from the Spacecraft stabilization and control system. Engine start may also be initiated by the crew in the Command Module by means of an override switch. The signal is delivered to the series mounted solenoid valves which open and allow fuel to pressurize the valve actuators. The actuator drives the propellant valves open by means of the interconnect link. Propellants flow through the open valves into the injector where they are distributed for injection into the combustion chamber.

Termination of engine operation shall be accomplished by removing the electrical signal from the solenoid valves, allowing the actuator pressure to exit overboard. Upon release of pressure from the actuator, the spring loaded piston moves the propellant valves to the closed position, shutting off propellant flow to the engine.

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Engine gimbal operation shall be accomplished by a redundant actuator system which receives its control signal from the Spacecraft stabilization and control system. Pilot override is also provided to allow crew operation in event of malfunction in the stabilization and control system.

3.4.2.5.1.1.3.4 Installation. - The rocket engine shall be mounted such that it is buried in the lower section of the Service Module. The engine shall be capable of being installed or removed as a package with the number of connect or disconnect points at a minimum.

3.4.2.5.1.1.3.5 Design Conditions. - The rocket engine design performance ratings shall be achieved with preset inlet conditions as follows:

Starting mode supply pressure = 178 ± 4 psia

Steady state supply pressure = 155 ± 4 psia

Propellant temperature = 70 F

3.4.2.5.1.1.3.6 Design Performance. - The rocket engine shall be designed to achieve the following performance ratings with propellants at inlet conditions as specified above.

	Initial	At 750 seconds
Thrust (lbs) -	21,935 ± 1%	21,935 + 10% - 1 %
Specific Impulse (seconds, minimum)	320	318.7

Mixture ratio - 2.0 ± 0.02

Expansion ratio - 60 : 1

Operating Life = 750 seconds minimum

3.4.2.5.1.1.3.7 Operating Conditions. - The engine shall be designed to operate throughout the operating life within the following conditions.

Normal Starting Mode. - Propellant supply pressure at the inlet to the engine valves will be 178 plus or minus 4 psia, prior to opening of the engine propellant valves.

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Steady State Operation. - Propellants, during steady state engine operation, shall be at 155 ± 4 psia.

Propellant Pressure Relationship. - During both steady state operation and the starting modes, the fuel and oxidizer pressures shall be within 2 psia of each other.

Propellant Supply Temperature. - The rocket engine shall be capable of operating with propellant at the following temperatures:

Fuel:	40 F to 160 F
Oxidizer:	30 F to 135 F

3.4.2.5.1.1.3.8 Performance Requirements.

Thrust Increase. - The rocket engine shall develop 90 percent rated thrust within 0.225 to 0.300 second after onset of electrical command signal. The start of transient impulse from run to run and from engine to engine shall be 1535 lb-sec ± 285 lb-sec from onset of electrical command to 90 percent rated thrust.

Thrust Decrease. - The rocket engine shall accomplish thrust decay to 10 percent rated thrust within 0.225 to 0.300 second after receipt of command signal. The rocket engine shutdown transients from run to run and from engine to engine shall be 4570 lb-sec ± 670 lb-sec from onset of electrical command signal to 10 percent rated thrust.

Engine Start. - The engine shall be capable of sustaining a minimum of fifty (50) start cycles during the operational life of 750 seconds.

Duty Cycle. - The rocket engine shall be capable of continuously operating for periods of 1.0 second minimum and 500.0 seconds maximum.

Frequency of Operation. - The rocket engine shall be capable of achieving 90 percent of rated thrust within 2.0 seconds after receiving a start signal. The engine shall be capable of satisfactory operation after a non-operating period of 45 days. The engine shall also be capable of satisfactory operation at any time during a 30-day period subsequent to non-firing functional checkout.

Chamber Cooling. - The rocket engine shall be designed to minimize the transmission of heat to the combustion chamber outer wall. The maximum temperature permissible on the outer wall, as a result of heat transmission from propellant combustion is 200 F.

3.4.2.5.1.1.3.9 "Abnormal" Conditions. - The rocket engine shall be designed for operation under several abnormal or "off design" conditions. While performance degradation is acceptable under these conditions, safety of operation cannot be compromised. The following paragraphs stipulate the conditions and acceptable performance.

Abnormal Starting Mode. - The rocket engine shall be capable of safely accomplishing a minimum of 10 starts with propellants furnished to the propellant valve inlets at a pressure of 240 psia.

Abnormal Steady State Operation. - The rocket engine shall be capable of safe steady state operation with the propellants furnished to the engine propellant valve inlets at pressures up to 215 psia. The engine shall be capable of continuous or intermittent operation under this condition.

Propellant Valve Failure. - The rocket engine shall be designed to provide safe starting and steady state operation with one set of fuel and oxidizer valves failed in the closed position. Thrust degradation of 4 percent is acceptable, however, a minimum specific impulse of 318.7 seconds must be maintained.

3.4.2.5.1.1.3.10 Performance Variation with Time. - The performance variation resulting from accumulated operating time shall be minimized. The acceptable variations in performance are as follows.

Thrust. - Thrust may increase throughout the operating life (750 seconds) by 10 percent maximum over the initial corrected design level. This limitation is established based on operation throughout the 750 second period at the most severe combination of operating conditions.

Specific Impulse. - The specific impulse shall not fall below 318.7 seconds at any time throughout the operating life (750 seconds) when operating with the most severe combination of operating conditions.

3.4.2.5.1.2 <u>Component Performance</u>. - A summary of the functional requirements for each category A, B and C component is outlined in Table III. Performance requirements which apply to each component category in general, are as follows:

3.4.2.5.1.2.1 <u>Fluid Compatibility</u>. -

<u>Category A Components</u>. - All category A (pressurization system) components shall be compatible with high-grade, oil-free commercial helium for long periods of exposure and intermittent exposures of short duration.

<u>Category B.1 Components</u>. - All category B.1 (oxidizer system) components shall be compatible with nitrogen tetroxide (N_2O_4) for long periods of exposure and intermittent exposures of short duration.

<u>Category B.2 Components</u>. - All category B.2 (fuel system) components shall be compatible with a mixture of 50 percent hydrazine (N_2H_4) and 50 percent unsymmetrical dimethylhydrazine (UDMH) for long periods of exposure and intermittent exposures of short duration.

<u>Category C Components</u>. - All category C (engine system) components shall be compatible with the fluids specified for category B.1 and B.2 components.

3.4.2.5.1.2.2 <u>Pressures</u>. - Helium source pressure for the pressurization system shall be approximately 4500 psig. Propellant shall be distributed to the engine at an approximate operating pressure of 155 psia.

3.4.2.5.1.2.3 <u>Leakage</u>. - All non-standard components shall be designed to perform, as required, with zero leakage.

3.4.2.5.1.2.4 <u>Electrical Requirements</u>. -

<u>Power Supply</u>. - All components, which are electrically actuated, shall operate from a power supply having the following characteristics:

Steady State Voltage	25-30 volts dc
Transient Voltage Limits	25-30 volts dc. Recovery time from one steady-state level to another upon load changes is less than 0.7 sec
Ripple Voltage	250 millivolts peak to peak maximum

Dielectric Strength. - All components, which are electrically actuated, shall withstand 1500 volts (RMS) at commercial frequency, for a period of one minute (at sea level), without evidence of insulation breakdown or flashover.

Insulation Resistance. - The insulation resistance for each component, which is electrically actuated, shall be a minimum of 100 megohms, when a potential of 500 volts dc is applied for a period of two minutes (at sea level).

Grounding. - All components which are electrically actuated shall not be internally grounded.

3.4.2.5.2 Reaction Control System. -

3.4.2.5.2.1 Function. - The function of the Service Module Reaction Control System shall be to provide three axis stabilization and control of the Spacecraft in the phases of flight defined below after Spacecraft launch.

Midcourse Velocity Correction. - The mission trajectory selected will influence the magnitude of midcourse velocity corrections required in the translunar or transearth phases. The Reaction System may supply minor velocity increments not supplied by the Service Propulsion System. The Service Module Reaction Control System's roll thrusters shall provide pitch and yaw maneuvering.

Lunar Vicinity Control. - The Service Module Reaction Control System shall provide the minor velocity increments and spacecraft stabilization and control during the lunar orbiting phase. The system shall provide ullage acceleration and roll control as necessary.

Preparation for Entry. - The system shall provide directional and attitude control of the Command Module and Service Module prior to Service Module jettison from the Command Module.

Separation. - The system shall provide a sustained translation acceleration of the Service Module along a two body axis after separation from the Command Module.

3.4.2.5.2.2 System Operation. - The complete Service Module Reaction Control System consists of four similar reaction control systems. The similar systems will be operated simultaneously during normal control operation. Each individual system consists of a pressurized helium storage and distribution system, an oxidizer storage and distribution system, a fuel storage and distribution system, and four rocket engine systems.

3.4.2.5.2.2.1 <u>Pressurized Helium System</u>. - The pressurized helium system shall be composed of the Category A components listed in Table IV and shown schematically in Figure 32. The helium supply shall be contained within one spherical tank which shall be cradle mounted to minimize detrimental tank flexures. A pressure regulation system consisting of two individual regulators connected in series shall be located downstream of the helium storage tank. The upstream regulator in the series shall reduce the high upstream pressure to a value slightly higher than the required downstream pressure. The second regulator in the series shall reduce the output of the first regulator to the required downstream pressure. If the upstream regulator fails open, the second regulator shall be capable of maintaining the required downstream pressure when subjected to the resulting increase in inlet pressure. Conversely, if the second regulator fails open the outlet pressure of the first regulator shall be within the permissible operating range of the propellant systems. A manually controlled, normally open solenoid valve shall be installed upstream of the pressure regulation system. The actuation of this valve shall provide the means of isolating the helium tank in the event of downstream leakage or component failure. The check valves downstream of the regulators shall prevent the propellants from entering the helium system in the event of failure of a propellant tank positive-expulsion device. The relief valve shall contain an integral helium filter and shall prevent an excessive pressure buildup in the propellant storage tanks. The vent valve shall provide means for venting the helium system downstream of the check valves during propellant servicing operations and helium depressurization operations.

3.4.2.5.2.2.2 <u>Oxidizer System</u>. - The oxidizer system shall be composed of the category B.1 components listed in Table IV and shown schematically in Figure 32. The fill valve shall provide the facility for servicing the oxidizer system during ground operations. The oxidizer supply shall be contained within a hemispherically domed cylindrical tank which shall be cradle mounted. The tank shall be equipped with a positive expulsion device. Pressurized helium from the helium system shall act on the opposite side of the positive expulsion device forcing the oxidizer through the oxidizer distribution system to the rocket engines at the required feed pressure. The

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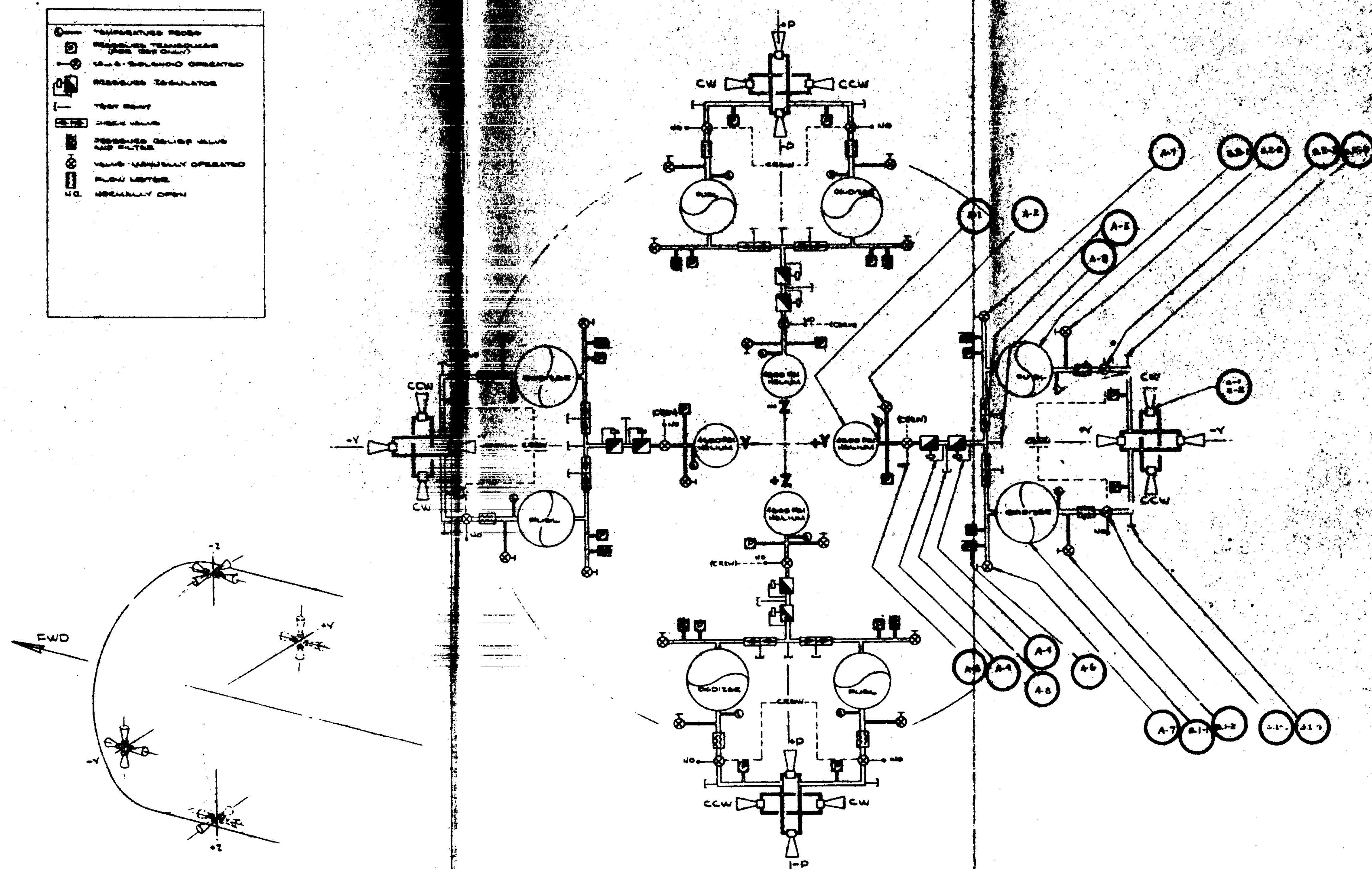

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Figure 32 Service Module Reaction Control System

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manually controlled, normally open solenoid valve shall provide the means of isolating the oxidizer storage tank from the rocket engines in the event of a leak in the oxidizer distribution system or a malfunction of a rocket engine.

3.4.2.5.2.2.3 Fuel System. - The fuel system shall be composed of the Category B.2 components listed in Table IV and shown schematically in Figure 32. The fill valve will provide the facility for servicing the fuel system during ground operations. The fuel supply shall be contained within a hemispherically domed cylindrical tank which shall be cradle mounted. The tank shall be equipped with a positive expulsion device.

Pressurized helium shall act on the opposite side of the positive expulsion device forcing the fuel through the fuel distribution system to the rocket engines at the required feed pressure. The manually controlled, normally open solenoid valve shall provide the means of isolating the fuel tank from the rocket engines in the event of a leak in the fuel distribution system or a malfunction of a rocket engine.

3.4.2.5.2.2.4 Rocket Engine System. - Each rocket engine system shall be composed of the Category C components listed in Table IV. Activation of the propellant feed system supplies propellant to the normally closed solenoid inlet valves mounted on the rocket engines. Electrical commands from the stabilization and control system open the oxidizer and fuel valves simultaneously. In the event of failure in the stabilization and control system, the pilot commands the engines on a separate manual circuit. The propellants, under system pressure, flow through their corresponding injector passages at high velocity, impinge in the combustion chamber and react exothermically. The expanded high temperature gases flow through the chambers and nozzle system producing the required thrust.

The service reaction control system shall employ 8 rocket engines for roll control, 4 engines for pitch control and 4 engines for yaw control. Control in each axis shall be supplied by pairs of engines arranged to provide rotation without translation. One engine of each pair shall be supplied by one of the independent propellant supply systems. Four engines shall be mounted in clusters of four on the external surface of the Service Module.

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3.4.2.5.2.2.4.1 Rocket Engine Performance. -

Rocket Engine. - The rocket engine shall be a pulse-modulated pressure fed, radiation cooled thrust generator.

Thrust. - The rocket engine shall develop a vacuum thrust during continuous operation of 100 pounds plus or minus 5 percent.

Thrust Transient Rate. - The rocket engine shall demonstrate a thrust buildup and thrust decay as described in Figure 26.

Specific Impulse. - The rocket engine shall achieve the following specific impulses during operation under vacuum conditions.

(a) Continuous Operation. - The rocket engine shall develop a specific impulse of 300 seconds when operating for periods in excess of one second.

(b) Pulse Mode Operation. - The rocket engine shall develop the specific impulse levels defined in Figure 27 when operating at pulse widths less than one second.

3.4.2.5.2.3 Component Performance. - A summary of the functional requirements for each category A, B and C component is outlined in Table IV. Performance requirements which apply to each component in general, are as follows:

3.4.2.5.2.3.1 Fluid Compatibility. -

3.4.2.5.2.3.1.1 Category A Components. - All Category A (pressurization system) components shall be compatible with high grade oil-free commercial helium for long periods of exposure and intermittent exposures of short duration.

3.4.2.5.2.3.1.2 Category B.1 Components. - All Category B.1 (oxidizer system components shall be compatible with nitrogen tetroxide (N_2O_4) for long periods of exposure and intermittent exposures of short duration.

3.4.2.5.2.3.1.3 Category B.2 Components. - All Category B.2 (fuel system) components shall be compatible with a mixture of 50 percent hydrazine (N_2H_4) and 50 percent unsymmetrical dimethylhydrazine (UDMH) for long periods of exposure and intermittent exposures of short duration.

3.4.2.5.2.3.1.4 Category C Components. - All Category C (engine system) components shall be compatible with the fluids specified for Category B.1 and B.2 components.

3.4.2.5.2.3.2 Pressures. - Helium source pressure for the pressurization system shall be approximately 4500 psig. Propellant shall be distributed to the engine at an approximate operating pressure of 170 psig.

3.4.2.5.2.3.3 Leakage. - All components shall be designed to perform as required with zero leakage.

3.4.2.5.2.3.4 Power Supply. - All components, which are electrically actuated, shall operate from a power supply having the following characteristics:

Steady State Voltage	25-30 volts dc
Transient Voltage Limits	25-30 volts dc. Recovery time from one steady-state level to another upon load changes is less than 0.7 sec.
Ripple Voltage	250 millivolts peak-to-peak maximum

Table IV. Service Reaction Control System Components

Category	Figure 32 Code No.	Component Title	Function
A	A-1	Vessel - Helium Pressure	Storage of high pressure helium
A	A-2	Valve - Helium Fill, Manual	Fill point during ground servicing operations
A	A-3	Valve - Helium, Solenoid Operated	Isolate the storage area in the event of a downstream failure
A	A-4	Regulator - Helium Pressure	Maintains the required constant downstream pressure
A	A-5	Check Valve Helium Pressure	Presents oxidizer and/or fuel from backing up into helium system
A	A-6	Relief Valve Helium Pressure	Presents overpressurization of the fuel and oxidizer system
A	A-7	Valve - Vent Helium Pressure	Depressurize low pressure side of helium system
A	A-8	Coupling - Check Helium Pressure	Provide pressure check points
B	B.1-1	Tank Oxidizer	Storage of nitrogen tetroxide (N_2O_4)
B	B.1-2	Valve - Oxidizer Fill, Manual	Fill point during ground servicing operation

Table IV. Service Reaction Control System Components (Cont.)

Category	Figure 32 Code No.	Component Title	Function
B	B.1-3	Valve - Oxidizer Solenoid Operated	Isolate storage area in the event of downstream failure
B	B.1-4	Coupling - Check Oxidizer	Provide pressure check point
B	B.2-1	Tank - Fuel	Storage of a 50/50 blend of UDMH and Hydrazine (N_2H_4)
B	B.2-2	Valve - Fuel Fill, Manual	Fill point during ground servicing operation
B	B.2-3	Valve - Fuel Solenoid Operated	Isolate storage area in the event of downstream failure
B	B.2-4	Coupling - Check Fuel	Provide pressure check point
C	C-1	Propellant Valves (2)	Control Propellant flow to the engine on demand
C	C-2	Thrust Chamber	Provides impulse for Spacecraft control

Grounding. - All components, which are electrically actuated, shall not be internally grounded.

Dielectric Strength. - All components, which are electrically actuated, shall withstand 1500 volts (RMS) at commercial frequency for a period of one minute (at sea level), without evidence of insulation breakdown or flashover.

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Insulation Resistance. - The insulation resistance for each component, which is electrically actuated, shall be a minimum of 100 megohms, when a potential of 500 volts dc is applied for a period of one minute (at sea level).

3.4.2.5.3 Communications and Instrumentation System. - Ref. paragraph 3.3.1.

3.4.2.5.4 Structural System. - Structural design considerations are defined in paragraph 3.4.2.6.

3.4.2.5.5 ECS. - The Service Module shall contain the following elements of the ECS.

Radiators

Thermal Load

Cryogenic Heat Exchanger

Cooling Water

Fuel Cell Water Supply

Water Fill Line

The functional operation of these elements shall be as defined in paragraph 3.3.3.

3.4.2.5.6 Electrical Power System. - The Service Module shall contain some components of the Electrical Power System. The components contained and their functions are defined in paragraph 3.3.2.

3.4.2.6 Structural Design. - The Service Module structure shall provide a mounting surface for all Service Module systems, help protect against the damaging effect of meteoroids, and withstand the flight loads expected during launch. The basic module structure shall consist of the following items.

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3.4.2.6.1 <u>Propulsion System Structure.</u> - The propulsion system structure shall contain the following items:

Helium Tank

Oxidizer Tanks

Fuel Tanks

Nozzle

3.4.2.6.2 <u>Service Module RCS Structure.</u> - The Service Module RCS structure shall contain the following items:

Helium Tanks

Oxidizer Tanks

Fuel Tanks

3.4.2.6.3 <u>Service Module ECS Structure.</u> - The following ECS elements are contained in the Service Module:

Radiators

Thermal Load

Cryogenic Heat Exchanger

Cooling Water Tanks

Fuel Cells

3.4.2.6.4 Internal Volume Identifications. - The Service Module shall consist of seven sextants consisting of a central sextant, cylindrical in shape and six partial pie shaped volumes identified as the upper, mid and lower right and left sextants. The tentative internal arrangement shall be as follows:

Sextant	Major Contents
Central	Propulsion Rocket Engine ECS Helium Tank
Upper Right Sextant	LH_2 Tank No. 1 LOX Tank No. 1 Fuel Cells (2) RCS He Tank RCS Fuel Tank RCS Oxidizer Tank DSIF Antenna Assembly
Mid-Right Sextant	ECS Oxidizer Tank
Lower Right Sextant	ECS Fuel Tank
Upper Left Sextant	ECS Oxidizer Tank
Mid-Left Sextant	ECS Fuel Tank
Lower Left Sextant	LH_2 Tank No. 2 LOX Tank No. 2 Fuel Cell RCS He Tank RCS Fuel Tank RCS Oxidizer Tank DSIF Antenna

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3.4.2.6.5 Loading Conditions. - Tank sizing shall be apportioned as follows. Figures are presented as percentages of usable propellant for both fuel and oxidizer except as noted.

Manufacturer's Tolerance	1.5 percent
Ullage (Oxidizer Only)	1.4 percent
Ullage (Fuel Only)	0.9 percent
Loading Tolerance	1.0 percent
Mixture Ratio Shift (O/F)	1.0 percent
Trap	0.5 percent

3.4.2.6.6 C.G. Management. - The design of the Service Module shall be such that no critical propellant utilization considerations will be necessary relative to center of gravity variations.

3.4.2.7 Interfaces. - Refer to paragraph 3.5.

3.4.2.8 Environment. - Refer to paragraph 3.10

3.4.3 Adapters. - The physical connection between the Service Module and the remaining parts of the Spacecraft and/or launch vehicle shall be by an adapter section. The Service Module shall be rigidly attached to the adapter section through a common interface. Some integrated systems functions may be exchanged through this interface until a separation operation is performed.

3.4.3.1 Performance Characteristics. - The performance of the Spacecraft Adapter shall be as specified herein.

3.4.3.2 Weight. - The adapter weight shall be the minimum consistent with design requirements and shall not exceed 5000 pounds.

3.4.3.3 Adapter Configuration. - Design of the adapters for the various missions shall require the same method of attachment, basic structure, and external geometry with respect to the Service Module.

3.4.3.4 Adapter Instrumentation. - The adapter instrumentation system shall consist of sensors and associate signal conditioners to provide structural integrity data as required.

3.4.3.5 Electrical. - The umbilical disconnects between the Service Module and the adapter and between the adapter and launch vehicle shall be used for checkout when the adapter is not mated to the adjacent module.

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3.5 Module Interfaces. -

3.5.1 Launch Escape System. -

3.5.1.1 Launch Escape System to Command Module. -

3.5.1.1.1 Electrical. -

A. Pitch motor initiation signal source

B. Launch escape motor initiation signal source

C. Tower jettison motor initiation signal source

D. Release mechanism initiation signal source

E. Instrumentation

3.5.1.1.2 Mechanical. - Four 10,000-12,000 lb preloaded latching studs.

3.5.2 Command Module. -

3.5.2.1 Command Module to Launch Escape System

3.5.2.1.1 Electrical. -

A. Pitch motor initiation circuit

B. Launch escape motor initiation circuit

C. Tower jettison motor initiation circuit

D. Release mechanism initiation circuit

E. Instrumentation sensors

3.5.2.1.2 Mechanical. - Four tower latch mechanisms

3.5.2.2 Command Module to Service Module. -

3.5.2.2.1 Electrical. -

A. Power System

1. +28 vdc source
2. 400 cycle load

B. Communication and Instrumentation System

1. DSIF Communication

a. Modulating Signals for DSIF transmission

1). Transmitter input

a). Characteristics

b). Information

(1). Voice

(2). Range

(3). Telemetry data

(4). Real time television

2). Receiver output

a). Characteristics

b). Information

(1). Voice

(2). Range

b. R-F level transmission

(to omni antenna)

1). Characteristics

a). 2200-2300 mc

b). Modulation

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(1). Phase

(2). Frequency

(3). Pulse coded

2). Information

a). Voice

b). Range

2. Monitoring system load

C. Stabilization and Control System

1. Input to engine firing circuit

2. Input to reaction jet firing circuits

3. Propulsion engine gimbal command circuit

D. In-Flight Test System

3.5.2.2.2 Mechanical.-

A. Structural Bond

1. Three latch mechanisms, 120° apart

2. Six bonded shear pad studs, located 60° apart

3. Seal

B. Umbilical Connection

1. Electrical

2. Environmental

3.5.2.2.3 Fluid.-

3.5.2.2.3.1 Hydraulic.-

A. Environmental Control System

1. Water Source

2. Glycol Load: space radiator

3.5.2.2.3.2 Pneumatic.-

A. Environmental Control System

1. Oxygen supply source

3.5.2.3 <u>Command Module to GSE</u>.-

3.5.2.3.1 <u>Fluid</u>.-

3.5.2.3.1.1 <u>Hydraulic</u>.-

A. Environmental Control System

3.5.2.3.1.2 <u>Pneumatic</u>.-

A. Environmental Control System

3.5.2.3.2 <u>Electromagnetic</u>.-

3.5.2.4 <u>Command Module to GOSS</u>.-

3.5.2.4.1 <u>Electromagnetic: Communication and Instrumentation System</u>.-

A Spacecraft - Surface - Spacecraft Communication

1. Near Earth (VHF)

a. Telemetry Reception (at GOSS):

1.) <u>Antenna</u>:

a). Type: Quad-helix

b). Gain: 18 db min.

c). Frequency bond: 225-260 mc

d). Polarization: left or right circular

e). Beam width: 20° conical

2.) <u>Receivers</u>:

a). Frequency Range; 225-260 mc

b). IF bandwidth: 300 kc at 3 db points

b. Voice Communication (two way)

3.) <u>Antennas</u>:

a). Type: Quad-helix

b). Gain: 18 db minimum

c). Frequency band: 250-300 mc

d). Polarization: Left or right circular

e). Beamwidth: 20° conical

4.) <u>Transmitter</u>:

a). Type: Double side-band amplitude modulation

b). Frequency range: 250-300 mc

c). Power output: 100 watts

5.) <u>Receiver</u>:

a). Type: Double side-band amplitude modulation

b). Frequency range: 250-300 mc

c). Sensitivity: 2 microvolt

c. Acquisition Aid

a). Type: active

b). Frequency: 225-260 mc

c). Antenna beamwidth: 20° minimum

d). Antenna polarization: Left or right circular

e). Tracking rate: 5° per second circular

f). Tracking error: 0.5° maximum

2. DSIF (with use of Command Module's omni antenna)

   a. Antennas

      1). GOSS station with TV capability

         a). Type: Parabolic

         b). Gain: 51.5 ± 0.5 db (85')

         c). Frequency band: 2100-2300 mc

      2). GOSS station without TV capability

         a). Type: Parabolic

         b). Gain: 44 db ± 1 db (28')

         c). Frequency band: 2100-2300 mc

   b. RF Pre-amplifiers

      1). GOSS stations with TV capability

         a). Type: Maser

         b). Gain: 20 db

         c). Frequency band: 2100-2300 mc

         d). Band width: = 20 mc (1 db pts)

      2). GOSS stations without TV Capability

         a). Type: Parametric

         b). Gain: 20 db

         c). Frequency band: 2100-2300 mc

         d). Band width: = 1 mc (1 db pts)

c. Transmitters

1). Modulation: FM/PM, PCM/PM

2). Output power: 10 KW

3). Frequency range: 2100-2300

4). Minimum Bandwidth: 5 mc

B. Tracking Equipment

1. Near Earth

a. C - Band Radar Interrogator

1). Frequency band: 5.4 to 5.9 mc

2). Coded interrogation of two or three pulses

2. Deep Space

C. Recovery Communication

1. HF Equipment Characteristics

a. Transmitter

1). Modulation: SSB-SC

2). Frequency range: 2-16 mc

3). Output power: 100 w, PEP

4). Frequency response: 300-3000 cps

5). Impedance at antenna terminals: 50 ohms

6). Minimum carrier suppression: 30 db

7). Maximum duty cycle: 5 out of 7 min

b. Receiver

1). Detection: SSB-SC

2). Frequency Range: 2-16 mc

3). R-F sensitivity: 1 microvolt

4). Frequency response: 300-3000 cps

5). Impedance at antenna terminals: 50 ohms

2. VHF Equipment Characteristics

a. Transmitter

1). Modulation: AM

2). Frequency range: 255-300 mc

3). RF power output: 3 watts average

4). Audio frequency response: 300-3000 cps

5). Impedance at antenna terminal: 50 ohms

6). Maximum duty cycle: 5 in 7 min

b. Receiver

1). Modulation: AM

2). Frequency range: 255-300 mc

3). RF power output: 3 watts average

4). Audio frequency response: 300-3000 cps

5). Impedance at antenna terminal: 50 ohms

3.5.3 Service Module

3.5.3.1 Service Module to Command Module

3.5.3.1.1 Electrical:

A. Power

1. +28 vdc load

2. 400 cycle source

B. Communication and Instrumentation System

1. Source for DSIF information to be transmitted

2. Omni-antenna for RF level communication during near earth operation

3. Monitoring system source

C. Stabilization and Control System:

1. Source of Reaction Control System signals

2. Source of propulsion engine firing sequence

D. In-Flight Test System

3.5.3.1.2 Mechanical

3.5.3.1.3 Fluid

3.5.3.1.3.1 Hydraulic

A. Environmental Control System

1. Water consumption load

2. Glycol inputs and outputs of the space radiator

3.5.3.1.3.2 Pneumatic:

A. Environmental Control System

1. Oxygen consumption load

3.5.3.2 Service Module to Vehicle Adapter

3.5.3.2.1 Electrical

3.5.3.2.2 Mechanical

3.5.3.2.3 Fluid

3.5.3.3 Service Module to GSE

3.5.3.3.1 Electrical

A. Power: 28 vdc source

B. Stabilization and Control System

C. Communication and Instrumentation System

3.5.3.3.2 Mechanical

3.5.3.3.3 Fluid

3.5.3.3.3.1 Hydraulic

A. Water supply source

3.5.3.3.3.2 Pneumatic

A. Oxygen supply source

B. Hydrogen supply source

3.5.3.3.4 Electromagnetic

3.5.3.4 Service Module to GOSS

3.5.3.4.1 Electromagnetic

A. DSIF (With use of Service Modules Parabolic antennas)

1. Antennas

a. GOSS station with TV capability

1). Type: Parabolic

2). Gain: 51.5 ± 0.5 db (85')

3). Frequency band: 2100-2300 mc

b. GOSS station without TV capability

1). Type: Parabolic

2). Gain: 44 db ± 1 db (28')

3). Frequency band - 2100-2300 mc

2. RF Pre-amplifiers

a. GOSS stations with TV capability

1). Type: Maser

2). Gain: 20 db

3). Frequency band: 2100-2300 mc

4). Bandwidth: ≥20 mc (1 db pts)

b. GOSS stations without TV capability

1). Type: Parametric

2). Gain: 20 db

3). Frequency band: 2100-2300 mc

4). Bandwidth: $\geq$1 mc (1 db pts)

3. Transmitters

a. Modulation: FM/PM, PCM/PM

b. Output power: 10 KW

c. Frequency range: 2100-2300

d. Minimum Bandwidth: 5 mc

3.5.4 <u>Vehicle Adapter</u>

3.5.4.1 Vehicle Adapter to Service Module

3.5.4.1.1 Electrical

A. Power

B. Stabilization and Control System

3.5.4.1.2 Mechanical

3.5.4.2 Vehicle Adapter to Vehicle

3.5.4.2.1 Electrical

A. Power

B. Stabilization and Control System

3.5.4.2.2 Mechanical

3.5.5 GSE

3.5.5.1 GSE to Command Module

3.5.5.1.1 Fluid

3.5.5.1.2 Electromagnetic

3.5.5.2 GSE to Service Module

3.5.5.2.1 Electrical

3.5.5.2.2 Fluid

3.5.5.2.3 Electromagnetic

3.5.6 GOSS

3.5.6.1 GOSS to Command Module

3.5.6.1.1 Electromagnetic

A. Spacecraft - Surface - Spacecraft Communication

1. Near Earth

a. Telemetry

1). VHF Transmitter

a). Modulation: FM

b). Frequency: 225-260 mc

c). Antenna: VHF Broad band antenna

d). Output: 10 watts minimum

b. Voice

1). VHF AM Transceiver

a). Modulation: AM CW, MCW

b). Frequency: 243-300 mc band

c). Antenna: VHF broad band antenna

d). Transmitter output: 10 watts

2. DSIF

a. Receiver

1). Modulation: PM

2). Frequency: 2110-2120 mc

3). Antenna: 2 kmc omni

b. Transmission

1). Modulation: PM or FM

2). Frequency: 2200-2300 mc

3). Output: 20 watt

4). Antenna: 2 kmc Parabolic

B. Tracking

1. Near Earth

a. C-band Transponder

1). Frequency band: 5.4 to 5.9 mc

2). Response: Single pulse

3). Output power: 2.5 KW peak power

2. Deep Space

C. Recovery

1. VHF Recovery Beacon

a. Frequency: 243 mc

b. Antenna: VHF Recovery Antenna

2. HF Transceiver

a. Frequencies: 2 to 10 mc band

b. Modulation: SSB

c. Antenna: HF Recovery Antenna

3.5.6.2 GOSS to Service Module

A. DSIF

1. Receiver

a. Modulation: PM

b. Frequency: 2110-2120

c. Antenna: 2 KC Parabolic

2. Transmission

a. Modulation: PM or FM

b. Frequency: 2200-2300

c. Power out: 20 watts

d. Antenna: 2 kmc Parabolic

[redacted]

3.6 Maintenance and Repair. -

3.7 Handling and Servicing Provisions. -

3.8 Pre-Launch Checkout. - Pre-launch checkout shall include the performance of all tests required to verify safety checks and functional capabilities for all Spacecraft systems for both off-pad and on-pad test operations.

3.8.1 Off-Pad Test Operations. - Off-pad test operations shall include those operations required for vehicle erection and alignment, connection of umbilicals, fuel lines, ground support equipment pertinent to Spacecraft pre-launch requirements, and establishing telemetry links for monitoring on-board systems. Tests performed off-pad shall include but not be limited to: water-glycol evaporator, cabin partial pressure control, space radiators, leak and pressure decay tests.

3.8.2 On-Pad Test Operation. - On-pad test operations shall include those systems confidence tests performed on the launch pad after the start of the launch countdown and prior to umbilical ejection and shall also include monitoring of Spacecraft systems and critical system parameters pertinent to crew safety, systems operation, combined systems tests and malfunction isolation. Systems tested on-pad shall include communication and instrumentation, environmental control system, super-critical gas storage, all Spacecraft systems pertinent to crew safety, and mission accomplishment.

3.9 General. -

3.9.1 Requirements for the Selection of Materials, Parts, and Processes. - Unless otherwise specified in the contract, materials, parts, and processes shall be selected with the following considerations:

3.9.1.1 Considerations for Selection. - The following considerations shall govern selection of materials, parts, and processes.

(a) Materials, parts and processes shall be suitable for the purpose intended. Safety, performance, reliability and, when applicable, maintainability of the item are of primary importance.

(b) Critical materials shall not be used except in those instances where their use is essential.

(c) Where suitable, materials and parts shall be of a kind and quality widely available in supply channels.

(d) When practicable, materials and parts shall be nonproprietary.

(e) When practicable, a choice among equally suitable materials and parts shall be provided.

(f) Single source items shall be avoided.

(g) When practicable, circuits shall be designed with a minimum of adjustable components.

3.9.1.2 Specification, Use of. - Materials, parts and processes shall be selected in the following order of preference provided coverage is suitable.

(a) Federal Specifications approved for use by NASA.

(b) Military Specifications and Standards (MIL, JAN or MS).

(c) Other Governmental specifications.

(d) Specifications released by nationally recognized associations, committees and technical societies.

3.9.1.3 Choice of Standard Materials, Parts, and Processes. - Preferred parts lists shall be used where applicable. Except as otherwise specified herein, when an applicable specification provides more than one grade, characteristic, or tolerance of a part or material; standard parts, materials, and processes of the lowest grades, broadest characteristics, and greatest tolerances which will enable the equipment to conform to performance and other requirements of the equipment specification shall be chosen and used in the equipment. However, when necessary to avoid delay in development or production, obvious waste of materials, or unnecessary use of production facilities; standard parts, materials, or processes of high grades, narrower characteristics, or smaller tolerances may be used. The requirements specified herein for the use of standard materials, parts, and processes shall not relieve the contractor of the responsibility of complying with all performance and other requirements set forth in the contract.

3.9.1.4 Non-Standard Parts, Materials, and Processes. - Non-standard parts, materials, and processes may be used when necessary to facilitate the design of the particular equipment. However, when such non-standard items are incorporated in the design they shall be documented as required by the contract.

3.9.1.4.1 New Parts, Materials and Processes. - New parts, materials, or processes developed under the contract may be used provided they are suitable for the purpose intended. Any new parts, materials or processes used shall be documented as required by the contract.

3.9.1.5 Miniaturization. - Miniaturization shall be accomplished to the greatest extent practicable commensurate with required function and performance of the system. Miniaturization shall be achieved by use of the smallest possible parts and by compact arrangement of the parts in assemblies. Miniaturization shall not be achieved by means that would sacrifice the reliability or performance of the equipment.

3.9.1.6 Flammable Materials, Use of. - Materials which may support combustion, or which are capable of causing an explosion, shall not be used.

3.9.1.7 Toxic Materials, Use of. - Materials which produce toxic effects or toxic substances under any combination of specified operating conditions shall be avoided where practicable.

3.9.1.8 Unstable Materials, Use of. - Materials which are known to emit or deposit corrosive substances, induce corrosion, or produce electrical leakage paths within an assembly, shall be avoided where practicable.

3.9.1.9 Fungus Resistant Materials, Use of. - Fungus inert materials shall be used to the greatest extent practicable. The materials listed in 3.9.1.9.1 are generally considered nonnutrients and shall be used in preference to those listed in 3.9.1.9.2.

3.9.1.9.1 Fungus Inert Materials. -

(a) Metals

(b) Ceramic (steatite, glass, glass bonded mica, etcetera)

(c) Mica

(d) Plastic using silica, glass, mica, or asbestos as a filler

(e) Teflon

(f) Polytetrafluoroethylene

(g) Cellulose acetate

(h) Polytrifluorocholoroethylene

(i) Polyvinyl chloride (co-polymers)

(j) Rubber (natural or synthetic)

(k) Silicone

(l) Polyethylene

(m) Polystyrene

(n) Polyamide

(o) Polyurethane

(p) Polyolefin

(q) Polysulfides

(r) Epoxys

(s) Polyesters

3.9.1.9.2 Fungus Nutrient Materials. -

(a) Cotton

(b) Linen

(c) Cellulose nitrate

(d) Regenerated cellulose

(e) Paper and cardboard

(f) Leather

(g) Cork

(h) Hair and felts

(i) Plastic materials using cotton, linen, paper, or wood-flour as a filler

3.9.1.9.3 Treatment of Fungus Nutrient Materials. - Fungus nutrient materials, when used, shall be hermetically sealed or treated and shall not adversely affect equipment performance or service life.

3.9.1.10 Materials, Use of. - All metals shall be of corrosive-resistant type or shall be suitably protected to resist corrosion during normal service life. Gold, silver, platinum, nickel, chromium, rhodium, palladium, titanium, cobalt, corrosion-resistant steel, tin, lead-tin alloys, tin alloys, Alclad aluminum, or sufficiently thick platings of these metals, may be used without additional protection or treatment.

3.9.1.10.1 Dissimilar Metals. - Dissimilar metals, as defined in Standard MS 33586, shall not be used in intimate contact unless suitably protected or coated to prevent electrolytic corrosion.

3.9.1.10.2 Electrical Conductivity. - Materials used in electronics or electrical connections shall have such characteristics that, during specified environmental conditions, there shall be no adverse effect upon the conductivity of the connections.

3.9.2 Documentation. - The contractor shall document his design to the extent specified in the contract. The contractor shall submit for NASA's approval all type I documents shown on figure 1.

3.9.3 Electromagnetic Interference Control. -

3.9.3.1 Spacecraft. - The requirements for interference control in the basic design of all electronic and electrical equipment, components, assemblies, and systems shall be as follows in accordance with Specification MC 999-0002.

3.9.3.1.1 Grounding. - The system design shall incorporate a single point ground (SPG) and balanced circuit shields for all circuits operating at less than 50 kc. Multiple Point Grounding (MPG) shall generally be employed for circuits operating above 50 kc, provided precautions are taken to insure that such grounding does not result in the occurrence of undesirable ground loops in the SPG system.

3.9.3.1.2 Returns. - The following returns shall be isolated from ground and routed separately. The return grounds are terminated at the single point ground in the Command Module.

(a) AC return (power)

(b) DC return

(c) Circuit return (isolated circuits)

(d) Shield return (audio)

(e) Chassis return

3.9.3.1.3 Bonding. -

3.9.3.1.3.1 Bonding Resistance. - Resistance measured between any electrical bond shall not exceed 0.080 ohm. A design goal of 0.010 ohm shall be established.

3.9.3.1.3.2 Bonding Impedance. - Impedance measured between any electrical or mechanical bond shall not exceed 0.080 ohm below one megacycle. A design goal of 0.010 ohm shall be established.

3.9.3.1.4 Shield Grounds. - Consideration shall be given to the use of multiple-grounded shields in unbalanced sensitive signal circuits of all equipment, and in all circuits operating above 50 kc. Single-grounded shields shall be considered for all differential inputs or balanced sensitive circuits below 50 kc.

3.9.3.1.5 Radio-Frequency Sensitive Circuits. - Equipment operated at frequencies above 50 kc may have signal and secondary power supply terminals mounted on the unit case. If such terminals are used, the following grounding and isolation provisions shall be required:

(a) Minimum dc resistance between all audio frequency (0 to 50 kc) input or output terminals or leads and the unit case shall be 1 megohm and a design goal of at least 100 megohms for high impedance sensitive signal circuits.

(b) Wire shields, including coaxial cable outer conductors for frequencies above 50 kc shall be multiple-grounded, continuous. Wire shields on audio-frequency input or output leads shall not be multiple-grounded.

(c) Secondary power supplies, such as static inverters and isolation transformers, shall be employed to isolate the ac power distribution system from the loads (units). DC isolation between the input and the output terminals of the secondary power supplies shall be 1 megohm or greater.

3.9.3.1.6 Audio-Frequency Sensitive Circuits. - The following design requirements shall apply to circuits sensitive to frequencies from 0 to 50 kc or above, as required by the contractor. Interconnecting line load termination impedance shall be at least 50 ohms unless otherwise specified by the contractor. Termination impedance ideally should be between 150 and 600 ohms except for special instrumentation circuits. The line source impedance is not restricted. Equipment interconnections shall be designed so that for each interconnecting circuit the signal return leads may be routed in the same bundle. Individual interconnecting circuits shall pass through connectors via adjacent pins. Internal wiring design shall employ wire twisting and wire routing as necessary to minimize magnetic field pick-up loops. Shields on airborne interconnecting wires used to exclude electrical fields shall be single-point grounded at the SPG point in the audio-frequency sensitive circuits. All shields shall be covered by a layer of insulation. The following grounding and isolation requirements shall be required:

(a) DC isolation between the unit case and any point on the signal or power circuit shall be 1 megohm or greater and a design goal of at least 100 megohms for high impedance sensitive signal circuits.

(b) Continuous shields on interconnecting wire shall not be multiple-grounded or connected in any way which creates a ground loop.

3.9.3.1.7 Power Distribution Considerations. - DC isolation between load equipment cases and all primary input power leads shall be 1 megohm or greater. All power leads carrying equal and opposite currents shall pass through adjacent connector pins. Each particular subsystem shall be analyzed to detect the presence of ground loops.

3.9.3.1.8 Transient Suppression. - Such devices as rectifiers, relays, solenoids, switching circuits and motors shall be so designed as to achieve freedom from interference and transients. External filtering is prohibited.

3.9.3.2 Associated Equipment. - Electromagnetic interference control shall be required for the launch vehicle, launch complex, ground support equipment, support buildings, and other associated equipment and shall conform to Specification MC 999-0002.

3.9.3.2.1 Space Vehicle. -

3.9.3.2.1.1 Ground Point. - A single-point ground (SPG) for the Spacecraft shall be established on the Command Module structure. Return ground leads of the Spacecraft shall be connected to the airframe by means of the SPG only.

3.9.3.2.1.2 Shielding. - Shields used to exclude electromagnetic interference at frequencies below 50 kc shall be grounded at one end only, specifically at the SPG.

3.9.3.2.2 Ground Equipment. -

3.9.3.2.2.1 Ground Point. - Ground power supplies connected to the Spacecraft shall be grounded at the Spacecraft SPG only.

3.9.3.2.2.2 Static Ground. - Static ground line shall be used to interconnect all stages.

3.9.3.2.2.3 Shield Grounds. - GSE shields connecting to the Apollo Spacecraft shall be held above ground in the GSE. Shields in the GSE not connected to Spacecraft shields shall be normally multiple-grounded for frequencies greater than 50 kilocycles and single-end grounded for frequencies below 50 kilocycles.

3.9.3.2.2.4 Bonding Impedance. - Bonding impedance from the launch pad foundations to the Spacecraft SPG shall be 8×10^{-3} ohms in the frequency range of 0 to 100 kc.

3.9.4 Special Tools. - The functional components of the Spacecraft and their attachments shall be designed such that the use of special tools for assembly, disassembly, installation, and service shall be kept to a practicable minimum.

3.9.5 Storage. - The complete Spacecraft airframe shall be designed to meet specification requirements after storage of three years when preserved, packaged and packed as specified herein.

3.9.6 Explosion. - The entire Spacecraft, including electronic systems and rocket motor ignitors, shall be designed to minimize the existence of fire hazards or explosive environment. The system shall be designed to prevent the emission of gaseous vapors which might contaminate the Command Module during any part of the mission operation. The fuel tanks mounted in the Command Module shall be compartmentized to prevent ignition in the event that leakage should occur. Where practicable, the various components shall be hermetically sealed or of explosion proof construction. The rocket motor squibs shall be capable of withstanding an electrical impulse of one ampere at one watt D.C. for five minutes without detonating, to meet the 95 percent confidence level specified in AFMTC Ordnance Standard With Regard to RF Radiation Hazards.

3.9.7 Nameplates and Product Marking. - The Spacecraft, and all assemblies, components, and parts shall be marked for identification in accordance with Standard MIL-STD-130.

3.9.8 Workmanship. - The Spacecraft, and all assemblies, components, and parts, shall be free of cracks, breaks, chips, bends, burrs, loose solder, loose attaching parts, loose electrical connections, and all other poor workmanship which would render the Spacecraft, and assemblies, components, and parts unsuitable for the purpose intended.

3.10 Environmental Conditions. -

3.10.1 General. - The Apollo Spacecraft and equipment shall be designed to meet the natural, (climatic), and induced environmental conditions which the equipment must withstand during transportation, handling, storage checkout and in operational use.

3.10.2 Ground Environments. -

3.10.2.1 Temperature. -

Conditions

Operating: 0° F to +160° F (Command Module Interior)

Non-operating: (Spacecraft Interior and Exterior)

Minimum: -55° F for 2 week period

Maximum: +160° F without solar radiation for two weeks or +125° F with 360 btu/ft^2/hr. solar radiation for two weeks.

Air Transportation: -65° F to +125° F without solar radiation for eight hour period.

3.10.2.2. Vibration. -

Non-operating: (Spacecraft)

Gross Weight	5-27.5 cps	27.5-52 cps	52-500 cps
wt. < 50 lbs.	±1.56 g	0.043 in. D. A.	±6.00 g
50 lbs < wt. < 1000 lbs.	±1.30 g	0.036 in. D. A.	±5.00 g
1000 lbs < wt.	±1.04 g	0.029 in. D. A.	±3.33 g

Non-operating: (Launch Escape Tower)

10 - 70 cps	±3 g
70 - 500 cps	±5 g

3.10.2.3. Shock. -

Non-operating: (Spacecraft)

Gross Weight	Shock Level	Time
wt. < 250 lbs.	30 g	11 ± 1 ms
250 lbs. < wt. < 500 lbs.	24 g	11 ± 1 ms
500 lbs. < wt. < 1000 lbs.	21 g	11 ± 1 ms
1000 lbs. < wt.	18 g	11 ± 1 ms
Launch Escape Tower Motor:	±10 g	11 ± 1 ms

3.10.2.4 Acoustics. -

Non-operating: (Spacecraft)

(To be determined)

3.10.2.5 Humidity. - The maximum humidity environment anticipated is 100 percent relative humidity including condensation of water or frost.

3.10.2.6 Sunshine. - Solar radiation at 360 btu/ft^2/hour for two weeks.

3.10.2.7 Rain and Freezing Rain. - Up to 4 inches/hour for one hour.

3.10.2.8 Snow. - Up to 1 inch/hour for one hour.

3.10.2.9 Sand and Dust. - As in desert and ocean beach areas, equivalent to 140 mesh silica flour with particle velocity up to 500 ft/minute.

3.10.2.10 Fungus. - As in tropical areas.

3.10.2.11 Salt Spray. - As near ocean beaches, equivalent to a 20 percent salt solution for a minimum of 50 hours.

3.10.2.12 Ozone. - Up to 3 year exposure to 0.05 parts/million concentration or 0.03 parts/million concentration for one year.

3.10.2.13 Hazardous Gases. - Explosion proofing requirements defined in MSFC drawing 10M01071.

3.10.2.14 Altitude. -

Operating: Facilities; 0 to 2000 ft.

Non-operating: Storage 0 to 2000 ft.

Transportation

Land. 0 to 6000 ft.

Air. 0 to 35,000 ft.

3.10.2.15 Electromagnetic Interference Control. - See paragraph 3.9.3.

3.10.2.16 Ground Winds. - Free standing for structural loading of launch vehicles (fueled or unfueled) 99.9 percent of strongest wind month at Cape Canaveral.

Vehicle Height	Steady Wind		Peak Wind	
Feet	Knots	Ft/Sec.	Knots	Ft/Sec.
10	23.0	38.7	32.2	54.5
30	28.8	48.5	40.3	68.0
60	33.6	56.8	47.0	79.4
100	37.5	63.4	52.5	88.6
200	42.6	71.9	59.6	100.8
300	46.0	77.9	64.4	108.9
400	48.3	81.8	67.6	114.2

3.10.3 Lift-Off Environments. -

Temperature

Command Module Interior 0°F to +150°F

Spacecraft Exterior -65°F to +300°F

3.10.3.1 Vibration. - Command Module and Service Module Interior equipment mounted to basic structure.

Frequency	Equipment less than 25 lbs.	25 lbs. or more
5 to 10 cps	0.30 inch D. A.	0.20 inch D. A.
10 to 20 cps	±1.60 g's	±1.0 g
20 to 63 cps	0.075 inch D. A.	0.05 inch D. A.
63 to 2000 cps	±15.0 g's	±11.0 g's

Random Vibration. -

(Equipment less than 25 lbs.) Random vibration whose spectral density is $0.02g^2/cps$ @ 5 cps, with a linear increase to $0.14g^2/cps$ @ 75 cps to 220 cps, then a linear decrease to $0.05g^2/cps$ @ 2000 cps.

(Equipment, 25 lbs. or more.) Random vibration whose spectral density is $0.01g^2/cps$ @ 5 cps with a linear increase to $0.125g^2/cps$ @ 90 cps to 220 cps then a linear decrease to 0.04 cps @ 2000 cps.

Launch Escape Tower Motor

5 to 10 cps	0.2 inches D. A.
10 to 60 cps	±3 g's
60 to 500 cps	±5 g's
500 to 2000 cps	±7 g's

3.10.3.2 Acoustics. -

Octive Band cps	External Level	Internal Levels	
		Service Module	Command Module
4.7 to 9.4	150 db	140 db	128 db
9.4 to 18.8	153 db	143 db	130 db
18.8 to 37.5	158 db	144 db	130 db
37.5 to 75	156 db	145 db	131 db
75 to 150	155 db	144 db	129 db
150 to 300	151 db	141 db	124 db
300 to 600	146 db	136 db	116 db

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Octive Band cps	External Level	Internal Levels Service Module	Command Module
600 to 1200	142 db	129 db	113 db
1200 to 2400	138 db	126 db	108 db
2400 to 4800	133 db	120 db	101 db
4800 to 9600	129 db	116 db	95 db
Overall	163 db	152 db	137 db

(For max. "q" abort 171 db overall)

3.10.3.3. Acceleration and Shock. -

Spacecraft 7.0 g's

Command Module Abort Condition 20.0 g's

Launch Escape System Motor. - 15 g steady state along major axis
20 g shock for 11 ms.

3.10.3.4 Humidity. - 0 to 100 percent relative humidity.

3.10.3.5 Sand and Dust. - As in desert and ocean beach areas.

3.10.3.6 Salt Spray. - As near ocean beaches.

3.10.3.7 Electromagnetic Interference Control. - See Paragraph 3.9.3.

3.10.3.8 Hazardous Gases. - Explosion proofing requirements defined in MSFC drawing 10M01071.

3.10.3.9 Ground Winds. - For Saturn (C-5) type vehicles at 99 percent probability levels for strongest wind month at Cape Canaveral at launching.

Vehicle Height	Steady Wind		Peak Wind	
Feet	Knots	Ft/Sec	Knots	Ft/Sec
10	18.4	31.1	25.8	42.9
30	22.9	38.7	32.1	54.2
60	26.4	44.6	36.9	62.3

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Vehicle Height	Steady Wind		Peak Wind	
Feet	Knots	Ft/Sec	Knots	Ft/Sec
100	29.3	49.5	41.0	69.3
200	33.6	56.8	47.0	79.4
300	36.5	61.7	51.1	86.3
400	38.7	65.4	54.2	91.5

3.10.3.10 Pressure Differential. - (Max. "q" conditions).

Fixed and deployable H. S.,	Burst = 3 psig	Crush = 8.3 psig
Pressure shell,	Burst = 9 psig	Crush = 8.3 psig

3.10.4 Boost Environments. -

3.10.4.1 Temperature. -

Command Module Interior	0°F to +150°F
Spacecraft Ext. Surface	-65°F to +300°F
Launch Escape Tower	-65°F to +800°F
Command Module Radome Area	To +1550°F for 110 seconds

3.10.4.2 Vibration. - (same as Lift-off conditions)

3.10.4.3 Acoustics. - (same as Lift-off conditions)

3.10.4.4 Acceleration and Shock. - (same as lift-off conditions) (shock transients to be determined)

3.10.4.5 Electromagnetic Interference Control. - See paragraph 3.9.3.

3.10.5 Earth Orbit Environments. -

3.10.5.1 Temperature. -

Command Module Interior	0°F to +150°F
Spacecraft Ext. Surface	-200°F to +300°F
Service Module Interior	-200°F to +300°F

3.10.5.2 Vibration. - (To be determined)

3.10.5.3 Acoustics. - Command Module Interior - overall sound pressure level 75 db.
Command Module Interior - crew sound interference 55 db.

3.10.5.4 Acceleration. - (To be determined)

3.10.5.5 Pressure. - Spacecraft exterior to 7.5×10^{-10} mm Hg.

3.10.5.6 Electromagnetic Interference Control. - See paragraph 3.9.3.

3.10.5.7 Meteoroids. - Flux considerations shall be based upon Whipple's Modified Distribution for Sporadic Meteoroids in the Vicinity of the Earth and Moon. Reference Figures 33, 34, and 35 for occurrence probability.

3.10.5.8 Radiation. - (Anticipated Spacecraft Exterior W/O Shielding) All systems, equipment, and components required for successful completion of the design missions shall experience no significant degradation in performance resulting directly or indirectly from a combined exposure anticipated according to nuclear radiation model as follows: (The exposures may be considered as separated in time, but the effects are to be considered from the total of the exposures as given).

Source	Particles	Corresponding Flux	Energy Greater Than
(Solar Cosmic Rays)	(Protons)	6.0×10^{11} Protons/CM^2*	100 MEV
		6.4×10^{13} Protons/CM^2*	5 MEV
(Gallactic Cosmic Rays)	(Protons)	1.86 Protons/CM^2/Sec	100 MEV
	(Alphas)	0.20 Alphas/CM^2/Sec	500 MEV
(Van Allen)	(Protons)	6.7×10^{7} Protons/CM^2**	100 MEV
	(Protons)	9.6×10^{7} Protons/CM^2**	40 MEV
	(Protons)	1.1×10^{12} Protons/CM^2**	20 MEV

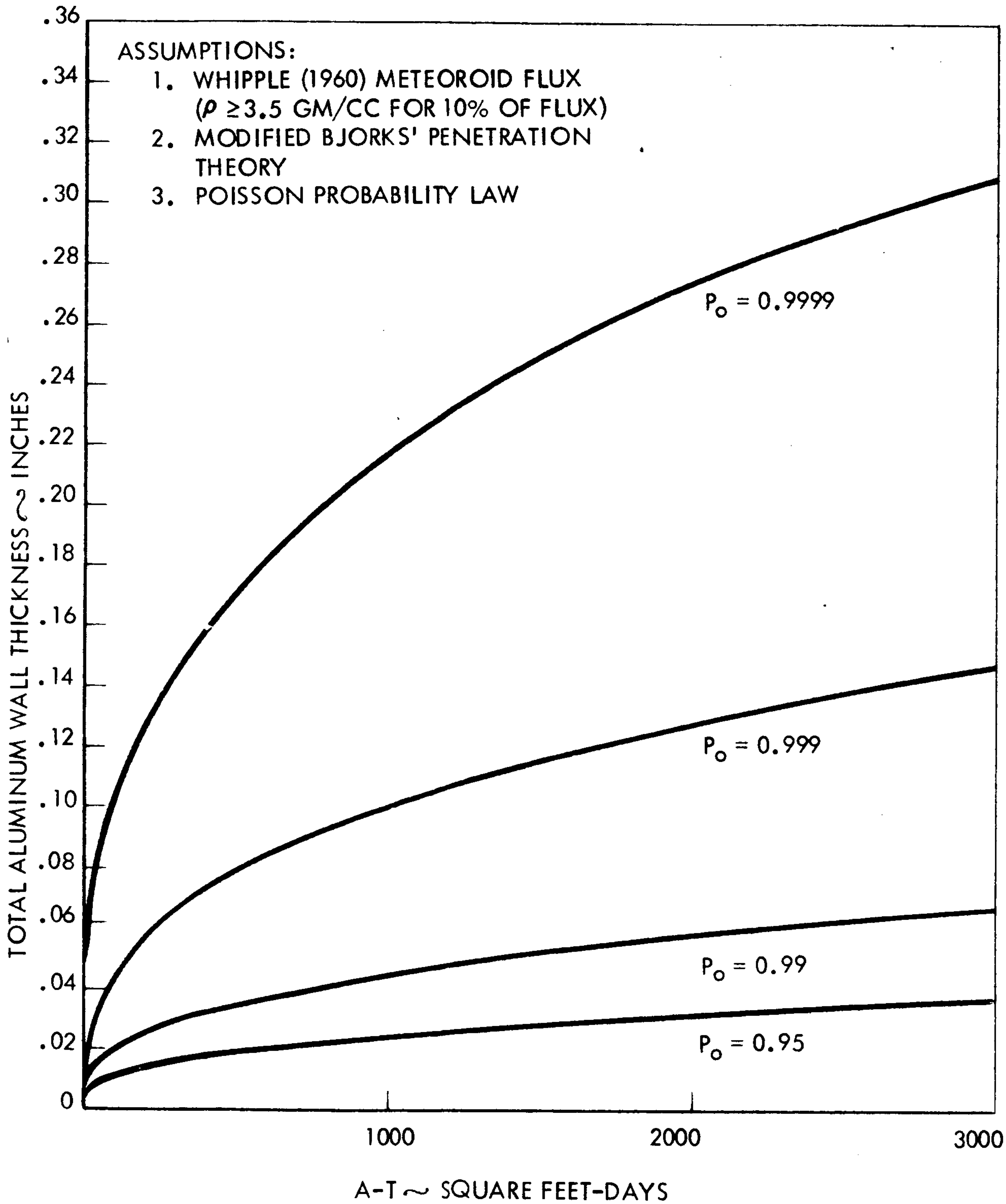

Figure 33. Aluminum Multi-Wall Thickness vs Target Area Time

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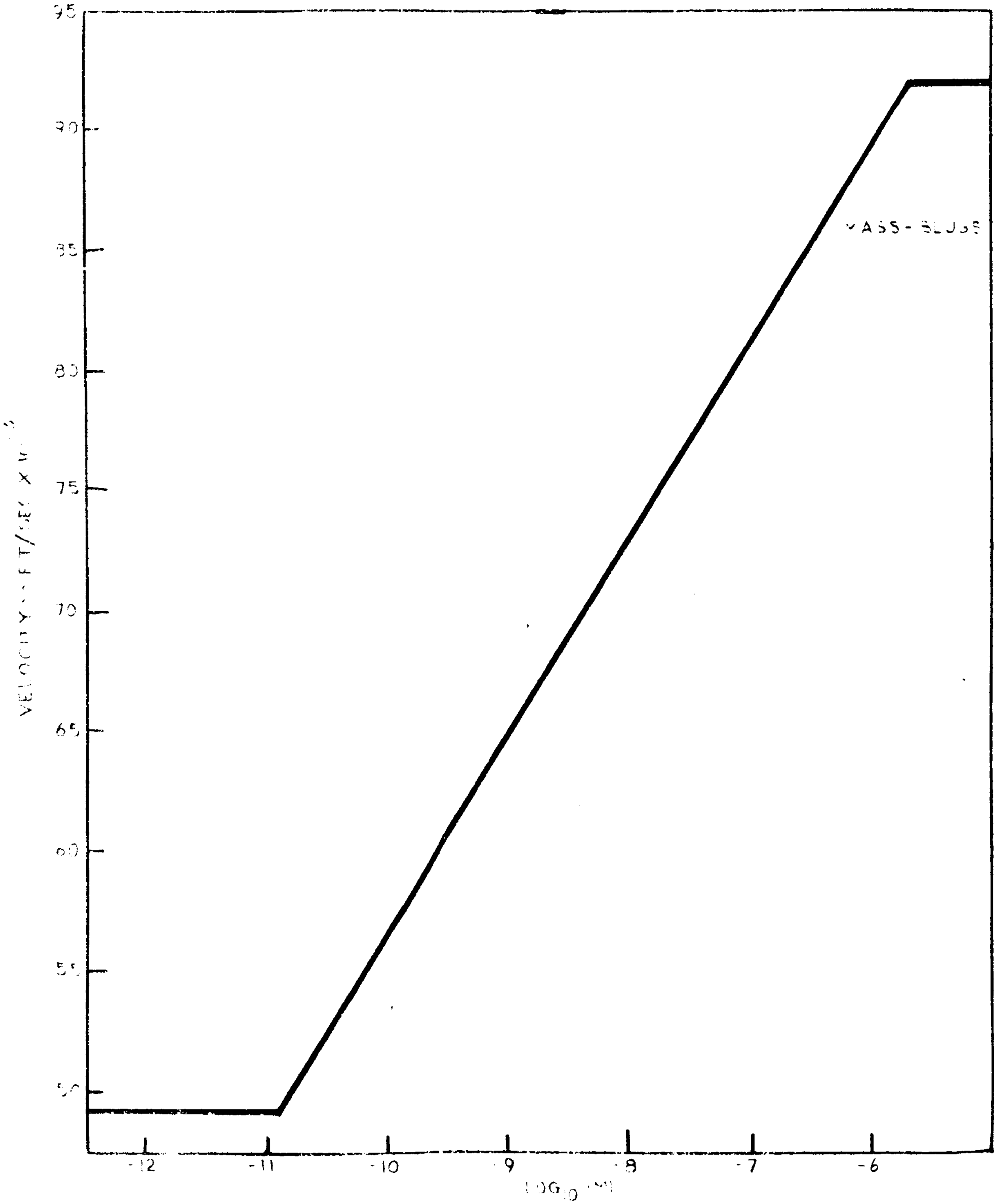

Figure 34 Whipples Mass Velocity Distribution

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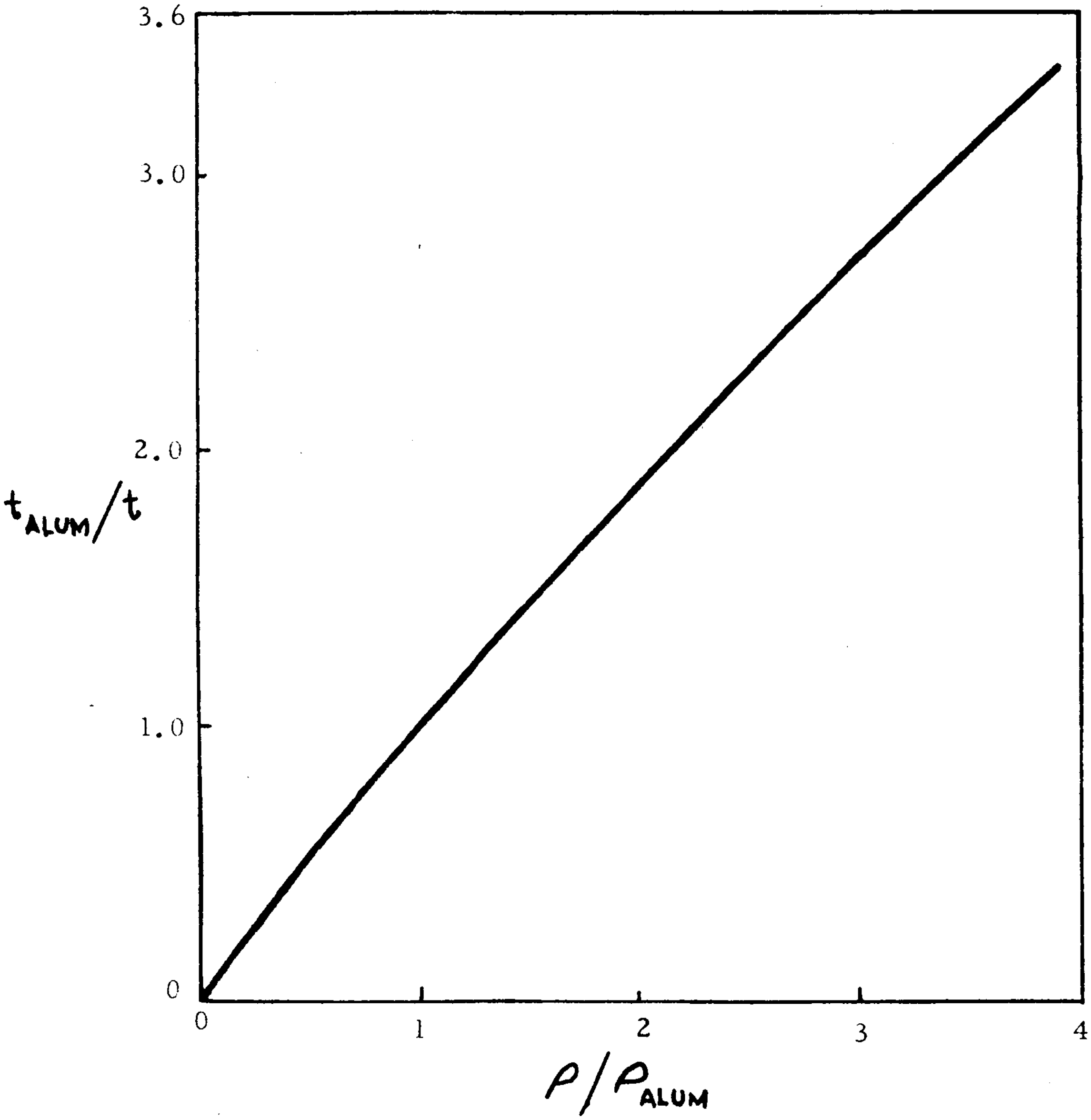

Figure 35 Meteoroid Shield Thickness Relative To Aluminum Shield Thickness

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Source	Particles	Corresponding Flux	Energy Greater Than
(Van Allen)	(Electrons)	1×10^{6} Electrons/CM^2**	5.0 MEV
		1×10^{9} Electrons/CM^2**	2.4 MEV
		1×10^{12} Electrons/CM^2**	0.04 MEV

*Flux computed on basis of May 10, 1959 event spectrum assuming 4 weeks total time in space beyond magnetosphere and assuming that 6.0×10^{11} protons/CM^2 represents the integral flux of the most intense event expected with a probability of 0.001 in four weeks.

**Integrated flux for four traversals.

3.10.6 Translunar Environments. -

3.10.6.1 Temperature. -

Command Module Interior	0°F to +150°F
Service Module Interior	-200°F to +300°F
Spacecraft Ext. Surface	-200°F to +300°F

3.10.6.2 Vibration. - (Same as Orbit)

3.10.6.3 Acoustics. - (Same as Orbit)

3.10.6.4 Acceleration. - (Same as Orbit)

3.10.6.5 Pressure. - 7.5×10^{-10} mm Hg (Exterior)

3.10.6.6 Electromagnetic Interference Control. - See paragraph 3.9.3.

3.10.6.7 Meteoroids. - (Same as Orbit)

3.10.6.8 Radiation. - (Same as Orbit)

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3.10.7 Transearth Environment. - (Same as translunar)

3.10.7.1 Earth Entry Environment. -

3.10.7.2 Temperature. -

Command Module Interior, $0^{\circ}F$ to $+150^{\circ}F$

Command Module Exterior, $-200^{\circ}F$ to $+6000^{\circ}F$ at heat shield

Command Module Interior Wall, to $+200^{\circ}F$

Parachute Compartment, to $+250^{\circ}F$

Command Module Backface, to $+600^{\circ}F$

3.10.7.3 Vibration. -

For steep 20 g entry - same as lift-off conditions

For shallow 10 g entry - one tenth lift-off conditions

3.10.7.4 Acoustics. -

For 20 g entry, exceeds 155 db overall for 110 sec. (Maximum at 20 g entry is 165 db)

For 10 g entry, 143 db for 1200 seconds

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3.10.7.5 Acceleration. -

Steep entry, 20 g

Shallow entry, 10 g

3.10.7.6 Shock. - (To be determined)

3.10.7.7 Pressure. - From 7.5×10^{-10}mm Hg to S. L. conditions.

3.10.7.8 Radio Interference. - See paragraph 3.9.3

3.10.8 Earth Landing Environment. -

3.10.8.1 Temperature. - Command Module Interior 0°F to +200°F

3.10.8.2 Shock. - (Command Module)

Crew Couch Design

Z-Z (eyeballs down) 20.0g's - system launch
Z-Z (eyeballs up) 15.0g's - system launch
Y-Y (eyeballs side) 15.0g's - system launch

(Nominal for C/M)

X-X (eyeballs in) 23.3g's
Z-Z (eyeballs down) 19.3g's
Z-Z (eyeballs up) 10.1g's
Y-Y (eyeballs side) 27.1g's

(Emergency for C/M) (Internal Equipment Support)

X-X (eyeballs in) 35.0g's
X-X (eyeballs out) 19.5g's
Z-Z (eyeballs down) 31.9g's
Z-Z (eyeballs up) 20.5g's
Y-Y (eyeballs side) 27.1g's

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3.10.8.3 Humidity. - To 100% R. H. with condensation

3.10.8.4 Sunshine. - Solar radiation at 360 $BTU/FT^2/Hr$.

3.10.8.5 Rain and Freezing Rain. - Up to 4 inches per hour

3.10.8.6 Salt Water and Salt Spray. - As on oceans

3.10.8.7 Sand and Dust. - As in desert and ocean beach areas

3.10.8.8 Waves. - Sea state 4 (less than 12 ft. waves)

3.10.8.9 Electromagnetic Interference Control. - (See Paragraph 3.9.3)

3.10.8.10 Ground Winds. - 30 knots at 10 feet

3.10.8.11 Entry Atmospheric Density Profile. - (See Paragraph 3.10.12.3)

3.10.8.12 Post Landing and Recovery. - Same as ground environments.

3.10.9 Mission Environments. -

3.10.9.1 Atmospheric Phase. -

3.10.9.1.1 Atmospheric Pressure, Density, and Temperature. - The altitude variation of atmospheric pressure, density, and temperature is given in "A Reference Atmosphere for Patrick AFB, Florida" by O. E. Smith in the NASA Annual TND-595, March 1961.

3.10.9.1.2 Wind. - The variation of surface wind with altitude for pre-launch and launch is given below. (Tables Nos. V and VI).

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Table V

Pre-Launch

Height (Ft)	Steady State Wind (Knots)	Peak Wind (Knots)
10	23.0	32.2
30	28.8	40.3
60	33.6	47.0
100	37.5	52.5
200	42.6	59.6
300	46.0	64.4
400	48.3	67.6

Table VI

Launch

Height (Ft)	Steady State Wind (Knots)		Peak Wind (Knots)	
	C-1	C-5	C-1	C-5
10	14.0	18.4	19.6	25.8
30	17.5	22.9	24.5	32.1
60	21.0	26.4	29.4	36.9
100	22.5	29.3	31.5	41.0
200	25.9	33.6	36.3	47.0
300	28.0	36.5	39.2	51.1
400	29.4	38.7	41.2	54.2

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3.10.9.2 Physical Constants for Moon. -

Model to be used:

Ratio of mass of earth to mass of moon: 81.45

$$\mu_m = GM_m: \quad 4.8938269 \times 10^{12} \ \frac{\text{meters}^3}{\text{sec}^2}$$

Radius of moon: 1738 kilometers

3.10.9.3 Entry, Landing, and Recovery Phase. -

3.10.9.3.1 Atmospheric Pressure, Density, and Temperature. - The altitude, seasonal, daily and latitude variation of pressure, density, and temperature will be as presented by the revised ICAO reference atmosphere. Because this reference is in the process of publication, the 1959 ARDC standard atmosphere will be used until the ICAO data is available. Certain sun, lunar, and planetary constants to be used are presented on page 165.

3.11 Ground Criteria. -

3.11.1 Factory Assembly and Checkout. -

3.11.1.1 Spacecraft Module Testing. -

3.11.1.1.1 Pressure Testing. - Where practical, pressure tests shall be performed using hydraulic pressure. When it is necessary to perform gaseous pressure checks, appropriate safety precautions shall be instituted to insure personnel safety.

3.11.1.1.2 Spacecraft Checkout Provisions. - Provisions shall be made for accomplishing all necessary checkout procedures without the necessity of making changes within the Spacecraft. Electrical and fluid systems shall not be disconnected to make test measurements if non-disruptive test points can be provided.

3.11.1.2 Combined Systems Checking. - Provisions shall be made for combined systems checkout as follows:

(a) At the component and subsystem level.

(b) At the major system level.

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Plant	M_s/M_p	ω	f	GM
Sun	1.	3.0050435×10^{-6}	0	$1.32715445 \times 10^{11}$
Mercury	6,120,000.		0	
Venus	406,645.	Synchronous	0	3.247695×10^{5}
Earth	332,488.		1/298.30	3.986032×10^{5}
Mars	3,088,000.	7.0882232×10^{-5}	1/191.8	4.297780×10^{4}
Jupiter	1,047.39	1.7734082×10^{-4}	1/15.2	1.267106×10^{8}
Saturn	3,500.	1.7055335×10^{-4}	1/10.2	
Uranus	22,869.	1.6135556×10^{-4}	1/14.	
Neptune	18,889.	1.1140400×10^{-4}	1/58.5	
Pluto	400,000.			

$$G = (6.668 \pm .0005) \times 10^{-8} \frac{cm^3}{sec^2\ gram}$$

$$M_e = (5.977 \pm .004) \times 10^{27} \text{ grams}$$

$$T = 86164.09054 \text{ seconds}$$

$$A_u = 1.49599 \times 10^{11} \text{ meters}$$

$$\frac{M_e}{M_m} = 81.375$$

Sun, Moon, and Planetary Constants

[redacted]

(c) Between systems of the same modules.

(d) Between systems of different modules.

3.11.1.3 Reliability Goals. - Design of the spacecraft and its systems, and all tooling, handling, and checkout equipment shall be such as to eliminate in-service equipment failures caused by errors in purchasing, packaging, handling, workmanship, inspection, and checkout. The elimination of these failures shall be considered as important as the elimination of failures resulting from inadequate design.

3.11.2 Transportation, Handling, and Storage. -

3.11.2.1 Hoisting, Jacking and Hauling. - All Spacecraft modules shall be designed to the following limit load factors:

	Vertical	Horizontal
Hoisting on Ship	2.67 g	1.0 g
Hoisting on Land	2.0	0
Jacking	2.0	.5

Transportation on Slow Moving Dolly - 5 MPH:

Vertical	Longitudinal*	Lateral*
2.0g	1.0 g	0 g
2.0	0	.75
1.0	1.0	.5

*With respect to dolly

The above factors include the effects of shock loading. Where possible the Spacecraft modules shall be cradled and handled such that ground shock do not exceed flight loads.

Transportation by Truck or Airplane - The Spacecraft modules shall be supported in mounts which limit the accelerations to the combination of 6.5 g longitudinal and 1 g transverse and to the combination of 4 g longitudinal and 2 g transverse, referenced to Spacecraft axes.

3.11.2.2 Transportability. - Each unit of GSE shall be capable of being lifted, transported by air, railroad or truck and reinstalled without hazard to the carrier, danger to the unit, or requiring special handling equipment.

3.11.2.3 Pressure Vessels in Transit. - Internal pressures during transportation shall be sufficient to prevent the occurrence of collapsing pressure.

3.11.2.4 Prelaunch Phase. -

3.11.2.4.1 Winds. - The laterally unsupported and unpressurized (free standing) space vehicle shall withstand the prelaunch wind profile shown in paragraph 3.10.2.16.

3.11.2.4.2 Load Factors. - A nominal limit longitudinal load factor of 1.0 g shall be considered at all times during the prelaunch phase. The transverse load factors shall be superimposed on the steady-state factor.

3.11.3 Prelaunch Operations. - The Spacecraft shall be designed to be compatible with the following requirements.

3.11.3.1 Erection and Mating. - Erection in the vertical checkout facility shall be compatible with the vehicle configuration and design loads.

3.11.3.2 Checkout and Servicing Requirements. - The Spacecraft modules, adapters, and stage simulators shall be designed such that system functional checks may be performed on any desired module.

3.11.3.3 Static Firing. - Static firings of the Service Propulsion System, or the reaction control system shall be accomplished at an attitude and gross weight such that the loads imposed upon the Spacecraft shall be within the structural limitation for flight loads.

3.11.3.4 Transport to Launch Pad. - The Spacecraft modules and adapter shall be capable of being transported to the launch pad in accordance with paragraph 3.11.2.

3.11.3.5 Launch Pad Operations. - The internal equipment and GSE shall be designed so that an integrated system check including radio noise and interference checks may be performed.

3.11.3.5.1 Countdown. - All systems shall be designed compatible with a countdown of two phases, consisting of prelaunch phase employing automatic sequencing, and a final phase under direct control of the crew.

3.11.3.5.2 Fueling. - The fueling operation shall be carried out with the Spacecraft in the launch position. The sequence shall be so as to make the prelaunch loads less critical than the flight loads.

3.11.3.5.3 Holds. - All Spacecraft and support systems shall be designed with the capability of withstanding hold periods of up to 12 hours. The hold period or succession of periods may occur at any time during the three-day countdown.

3.11.3.5.4 Scrubbed Missions. - An automatic shutdown sequence shall be made available for operation by the launch control center or by the crew. This sequence shall shut down the systems in such a manner as to insure complete safety to the crew and no damage to the equipment. Unloading sequences shall impose loads less critical than the flight loads.

3.11.3.6 Human Factors Considerations. - The ground crew and Spacecraft checkout crews shall be a selected group of personnel that will be trained in the operation of checkout procedures, and servicing and maintenance of equipment. Consideration shall be given to desirable levels of intelligence, motivation and physical performance.

4. QUALITY ASSURANCE PROVISIONS

4.1 General quality assurance provisions. - The contractor is responsible for the performance of all inspection requirements as specified herein. Except as otherwise specified, the contractor may utilize his own or any other inspection facilities and services acceptable to the Government. Inspection records of the examinations and tests shall be kept complete and available to the Government as specified in the contract or order. The Government reserves the right to perform any of the inspections set forth in the specification where such inspections are deemed necessary to assure supplies and services conform to perscribed requirements.

4.2 Contractor's quality assurance program. - The contractor shall establish a quality assurance program in accordance with Publication NCP-200-2. Inspections and tests to determine conformance of the system to contract and specification requirements shall be conducted prior to submission of the article to NASA for acceptance.

4.3 Examination. - Each assembly and major component submitted for acceptance shall be subjected to a visual examination to determine conformance to materials, design, construction, dimensions, weight, color and finish, product marking, and workmanship.

4.3.1 Components. - The inspector shall ascertain that prior to assembly, all parts, components, and assemblies procured under separate specifications or drawings have been inspected, tested, and accepted in accordance with their respective specifications or drawings.

4.4 Tests. - Each assembly and major component submitted for acceptance shall be subjected to performance tests as specified in their detail specification.

5. PREPARATION FOR DELIVERY

5.1 Preservation, packaging, and packing. - Preservation, packaging, and packing shall be accordance with the contractor's procedure provided the procedure will assure adequate protection in accordance with delivery modes, destinations, and anticipated storage periods.

6. NOTES

6.1 Definitions. -

6.2 Sign Conventions. - For the purpose of this document, and in the interest of establishing common reference points, the general sign conventions used herein are defined as follows:

6.2.1 Reference Axes. - The reference axes of the Spacecraft shall be orthogonal and shall be identified as shown on Figure 36.

6.2.1.1 X-Axis. - The X-Axis shall be parallel to the nominal launch axis of the Spacecraft, and shall be positive in the direction of initial flight.

6.2.1.2 Y-Axis. - The Y-Axis shall be normal to the X-Axis, and positive to the right of a crewman when the crewman is facing toward positive "X" in the launch position.

6.2.1.3 Z-Axis. - The Z-Axis shall be normal to both the X- and Y-Axis and shall be positive in the direction of the crewman's feet.

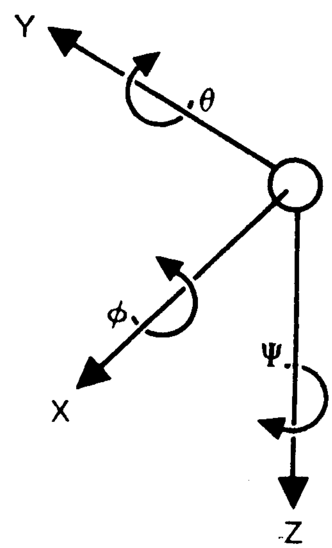
6.2.2 Nomenclature. - For the purpose of this document, and in the interest of establishing common terminology, nomenclature and terms used herein are defined as follows:

6.2.2.1 Absorptance. - Ratio of absorbed radiant energy to the incident radiation. In this specification, absorptance is over the solar spectrum.

6.2.2.2 Advisory Light. - A placard light used to inform the crew member of a safe or normal configuration, equipment or system status, operation of essential equipment, or to attract attention and impart information of a routine nature.

6.2.2.3 Albedo. - Ratio of radiant energy reflected from a planet or satellite to that received by it. A dimensionless decimal quantity equal to or less than 1.

Positive direction of axes and angles (forces and moments) are shown by arrows. (When launch vehicle is at a launch angle of 90°, the positive "X" direction is vertically upwards.)



Axis		Moment About Axis		
Designation	Symbol	Designation	Symbol	Positive Direction
Longitudinal	X	Rolling	L	Y ⟶ Z
Lateral	Y	Pitching	M	Z ⟶ X
Normal	Z	Yawing	N	X ⟶ Y

Force	Angle		Velocities	
(Parallel to Axis Symbol)	Designation	Symbol	Linear (Components along Axis)	Angular
X	Roll	ϕ	U	p
Y	Pitch	θ	V	q
Z	Yaw	ψ	W	r

Figure 36. Reference Axes, Spacecraft

6.2.2.4 Areas of Illumination Control. - This term shall designate a partial instrument panel area, primary duty station area or secondary duty station area illuminated by a given set of lumenaires.

6.2.2.5 Automatic Control. - Any device whose function is to activate some mechanism or equipment without the aid of human control.

6.2.2.6 Backpack. - A portable life support system providing atmospheric and thermal control.

6.2.2.7 Biomedical Instruments. - The instrumentation associated with pickup, recording and transmitting physiological data.

6.2.2.8 Breadboard Model. - An assembly of preliminary circuits and parts employed to prove the feasibility of a device, circuits, equipment, system, or principle in rough or breadboard form, without regard to the eventual overall design or form of parts.

6.2.2.9 Caution Light. - A placard light used to inform the crew members of an impending dangerous condition requiring attention but not necessarily immediate corrective action, such as "electrical compartment overheated" or "cabin pressure low."

6.2.2.10 C-Band. - 4000 to 7000 mc.

6.2.2.11 Ceiling Area. - That area above the intersection of the vertical wall and sloping wall or above consoles with the capsule in the launch position. This area begins approximately 30 inches above the capsule floor and side wall intersection.

6.2.2.12 Clinical Devices. - Devices for making basic physiological measurements indicative of crew well-being, such as thermometers and blood pressure measuring devices.

6.2.2.13 Combined Stresses. - Combined stresses are those stresses caused by simultaneous action of all factors, e.g., direct loading, thermal stresses.

6.2.2.14 Creep and Fatigue. -

6.2.2.15 Command Functions. - The manual and automatic control of the vehicle during all phases of the mission; the selection, impelmentation, and monitoring of modes of navigation and guidance; and monitoring and control of key areas of all systems during time critical periods.

6.2.2.16 Command Reaction Control System. - A control system which provides thrust vectors for three-axis control of the Command Module. This terminology does not include the associated Navigation and Guidance System.

6.2.2.17 Contrast Ratio. - Contrast ratio is $C = \frac{B_2 - B_1}{B_1}$ where B_1 is the brightness of the background and B_2 is the brightness of the lettering, numbering, or markings.

6.2.2.18 Console. - As used in ground support equipment nomenclature, a console is a short sloping front frame upon which standard 19-inch or 24-inch panels are mounted to support instrument indicators and controls. As used in the Spacecraft (Command Module) a console is any front frame upon which are mounted controls and displays, except for the main instrument panel.

6.2.2.19 Crew Environment Requirements. - Environmental needs to sustain crew life and provide for comfort and efficiency in work, rest, and recreation.

6.2.2.20 Crew Performance. - The capability of a crew member to perform a required or an assigned task with an acceptable degree of proficiency.

6.2.2.21 Crew Requirements. - Requirements and provisions necessary to maintain crew health and well-being and to assure effective crew performance.

6.2.2.22 Crew Safety. - Maintenance of the well-being of the crew within certain specified limits and probabilities.

6.2.2.23 Crew Status. - The state of well-being, including physical and mental health of the crew, as indicated by measurable physiological changes.

6.2.2.24 Crew Systems. - The organization, function, and interrelationship of crew members in regard to their responsibilities and performance as a subsystem in support of the Spacecraft in performing the specified missions.

The arrangement of the Command Module interior, the crew life support and survival equipment, and miscellaneous comfort items also fall within this category.

6.2.2.25 Design Burst Pressure. - The maximum limit pressure multiplied by the appropriate ultimate factor of safety.

6.2.2.26 Design Ultimate Load. - The limit load multiplied by the ultimate factor of safety.

6.2.2.27 Design Yield Load. - The limit load multiplied by the yield factor of safety.

6.2.2.28 Design Yield Pressure. - The proof pressure multiplied by the appropriate factor of safety.

6.2.2.29 Display. - Any device which exhibits, presents or imparts information to the operator or crew member in any manner. This may be an indicator, or groups of indicators, a label, or a gross machine transmitting motion, vibration or acoustic information to operator or crew member.

6.2.2.30 DSIF. - Deep Space Instrumentation Facility.

6.2.2.31 Electrical Heat Load. - The heat load on the environmental control system resulting from electrical and electronic heat dissipation.

6.2.2.32 Emergency Controls. - Those controls used during an emergency situation where proper operation is mandatory and where inadvertent operation could result in component destruction or crew member injury. Examples are destruct switches, fire extinguisher switches, and ejection controls.

6.2.2.33 Emergency Limits. - The limits beyond which there is high probability of permanent injury, death, or incapacity to such an extent that the crew would not perform well enough to survive.

6.2.2.34 Emittance. - Ratio of total emissive power of a body to emissive power of a black body at the same temperature. In this specification, emittance refers to total emissive power.

6.2.2.35 Environmental Heat Load. - The heat load on the environmental control systems which results from heat transfer through the Command Module wall.

6.2.2.36 Factor of Safety. - The ratio of the design load on a structure to the limit load.

6.2.2.37 Ground Support Equipment (GSE). - Ground support equipment is the equipment required to inspect, test, adjust, calibrate, appraise, gage, measure, repair, overhaul, assemble, disassemble, transport, safeguard, record, store, actuate, service, maintain, launch, and otherwise support an end article. Ground support equipment shall include ground-based training equipment, simulation devices, and auxiliary power devices.

6.2.2.38 HF. - High frequency, 3-30 mc.

6.2.2.39 Human Engineering. - The determination of man's capabilities and limitations as they relate to the mission environment conditions, crew station provisions, and other equipment he will use, and the application of this knowledge to the planning and design of complete systems, support systems, and operational support equipment so that the reliability and efficiency of the resultant man-machine combinations will be increased.

6.2.2.40 Human Factors. - The scientific determination of facts about human behavior, the development of systematic methods for considering man in the design of systems, and the application of these facts and methods throughout design. It includes the development and application of procedures and principles for the design of work spaces, equipment, human tasks, training, and human environments.

6.2.2.41 Human Waste. - Urine, feces, sputum, nasal discharge, carbon dioxide and other gases.

6.2.2.42 Indicator. - Any discrete display device providing a crew member with a qualitative or quantitative indication. This may be a counter, light, mechanical lever movement, or other device which changes color, illumination, or position as a function of another mechanism or condition.

6.2.2.43 Indicator Light. - A signal light assembly having no markings on the illuminated surface and used to present equipment status or similar information to crew members. The light is not intended as an attention attractor and is usually used in the secondary duty area.

6.2.2.44 Instrument. - A device for detecting and measuring some physical phenomenon. The output of an instrument may be an indication or a signal to one or more recording devices, or to one or more telemetering devices.

6.2.2.45 Instrument Panel. - A panel upon which are mounted instruments and their associated displays and controls, such as switches and adjustment knobs for ready scanning and operation by an operator or crew member.

6.2.2.46 Integral Illumination. - That illumination which originates within a display or indicator device. An integrally illuminated display contains its own light sources.

6.2.2.47 Interchangeability. - Interchangeability applies to all completely finished electrical or mechanical assemblies, components, and parts which shall be capable of being readily installed, removed, or replaced without alteration; misalignment or damage to parts being installed or to adjoining parts. No fabrication operations such as cutting, filing, drilling, reaming, hammering, bending, prying or forcing shall be required for the installation procedure; however, adjustment of variable resistors, trimming capacitors, etc, may be made.

6.2.2.48 Life Support System. - The complete system necessary for sustaining life in an alien environment, including atmosphere control, thermal control, food and water provision, waste disposal and sanitation, and radiation protection.

6.2.2.49 Limit Load. - Maximum calculated load to which the structure will be subjected under specified conditions of operation.

6.2.2.50 Limit Pressure. - Maximum pressure to which the structure will be subjected under specified conditions of operation. For Spacecraft propellant tank design, maximum limit pressure is the maximum relief valve pressure plus hydrostatic head (if applicable) and the minimum limit pressure is the minimum vapor pressure of the propellant under specified conditions of operation plus hydrostatic head (if applicable).

6.2.2.51 Load Factor. - A body load parameter equal to the ratio of net force applied to the body in a given direction divided by the weight of the body.

6.2.2.52 Lunar Sub-Solar. - Lunar noon or that point in time of the lunar light period when the sun's rays are perpendicular to the lunar surface at the equator.

6.2.2.53 Main Instrument Panel. - That instrument panel or collection of instrument panels upon which are mounted the primary attitude and condition indicators for the Command Module.

6.2.2.54 Maintainability. - A quality of the combined features of material design and installation which permits or enhances the accomplishment of maintenance by personnel of average skill, under the environmental conditions in which the maintenance will be performed. It includes reparability and serviceability, and is a function of the rapidity and ease with which maintenance operations can be performed to avert malfunctions or correct them as they occur.

6.2.2.55 Maintenance. - The servicing, repair, care, modification, or other action taken to keep material or equipment in, or restore it to, such condition as to meet programmed operational requirements.

6.2.2.56 Man Heat Load. - The heat load, both sensible and latent, rejected by the crew members to either the environmental control system or back pack system.

6.2.2.57 Manual Limits. - Limits within which the crew's environment shall be maintained during normal operations.

6.2.2.58 Margin of Safety. - The percentage by which the allowable load or stress exceeds the design load or stress.

$$\text{M.S.} = \frac{\text{Allowable Load or Stress}}{\text{Design Load or Stress}} - 1.0$$

6.2.2.59 Master Caution Light. - A placard light used to inform the crew members that one of a number of caution lights has been actuated. It is located at the primary duty station.

6.2.2.60 Maximum Relief Valve Pressure. - The maximum pressure which a relief valve will permit to exist in the system.

6.2.2.61 Metabolic Requirements. - Human energy exchange needs, such as oxygen, water and food consumption, and carbon dioxide and leak outputs.

6.2.2.62 MEV. - Million electron volts.

6.2.2.63 Minimum Relief Valve Pressure. - The lowest pressure at which the valve starts to open. This is the minimum pressure to be used for pressure-fed propellant tanks and systems.

6.2.2.64 Natural Environment. - The sum total of the external influences, such as conditions of the atmosphere, gravity, radiations, to which the crew will be exposed throughout the entire course of the mission.

6.2.2.65 Non-Human Waste. - Food remnants, shaving, dental cleansing, and body cleansing wastes, such as tissues and towels and paste.

6.2.2.66 Non-Stressed Limits. - The environmental limits to which the crew may be subjected for extended periods of time, such as orbit, lunar transit, and periods subsequent to normal landings.

6.2.2.67 Noxious Gases. - Gases which are potentially toxic to the crew, including CO_2, CO, H_2S, SO_2, methane, indole, skatole, mercaptans and ozone.

6.2.2.68 NRZ. - Non-return to zero.

6.2.2.69 Operating Pressure. - Operating pressure, or working pressure, is the nominal pressure to which components are subjected in service under steady-state conditions. Maximum operating pressure is the operating pressure plus tolerance.

6.2.2.70 PAM. - Pulse Amplitude Modulation.

6.2.2.71 PCM. - Pulse Code Modulation.

6.2.2.72 PEP. - Peak Envelope Power.

[redacted]

6.2.2.73 Percentile. - A statistical term denoting the hypothetical person within a population who has a dimension such that a certain percent of the population has smaller dimensions. Thus: the "10th percentile man" has all body dimensions such that 10 percent of the men considered have smaller dimensions, and 90 percent have larger dimensions.

6.2.2.74 PTT. - Push to talk.

6.2.2.75 Placard Light. - A signal light assembly having markings on the illuminated surface and used to present information of prime importance to the crew. Placard lights are intended as attention attracting devices and are usually used in the primary duty station.

6.2.2.76 PM. - Phase Modulation.

6.2.2.77 Potable Water. - Water of sufficient purity for human consumption.

6.2.2.78 Pressurization System. - A system which stores, regulates, and distributes helium for propellant tank pressurization, feed-out, and pneumatic valve operation.

6.2.2.79 Primary Duty Station. - Crew position for operation and monitoring of primary displays, controls and support systems.

6.2.2.80 Proof Pressure. - Equal to maximum limit pressure multiplied by the appropriate factor of safety and is the reference pressure for establishing acceptance test pressure levels.

6.2.2.81 Propellant System. - A system which stores and distributes the propellants to the associated engines and/or reaction jets.

6.2.2.82 Radiation Dosimeters. - Devices for recording amount of radiation exposure.

6.2.2.83 Radiobiological Terms. -

(a) Roentgen: That amount of X or gamma radiation whose energy per photon is less than 3 Mev which produces in 0.001293 grams of dry air, under conditions of electronic equilibrium e.g., one electrostatic unit of ionization charge of either sign.

[redacted]

(b) Absorbed Dose: The amount of energy absorbed in a unit mass of any material, the unit of absorbed dose is the RAD (Radiation Absorbed Dose) which is defined as 100 ergs per gram of any irradiated material.

(c) RBE (Relative Biological Effectiveness): The ratio of the RAD dose of x rays to the RAD dose of any other radiation required to produce an identical biological effect in a particular organ or tissue.

(d) REM (Roentgen Equivalent Man): This is used to express human biological doses as a result of exposure to one or many types of ionizing radiation. The dose in REMS is equal to the absorbed dose in RAD's times the RBE factor of the type of radiation being absorbed. Thus the REM is the unit of RBE dose.

(e) Radiation Dose: Space ionizing radiation exposure considered acceptable by NASA tabulated below.

(f) Blood Forming Organs: Lymph node, bone marrow, spleen and other tissue related to blood component formation.

Table of Space Ionizing Radiation Exposure Dose Limits

Critical Organ	Average yearly dose (rads)	Maximum permissible single acute emergency exposure (rads)	Location of dose points*
Skin of whole body	125	500	0.07 mm depth from surface of cylinder 2 at highest dose rate point along eyeline.
Blood forming organs	50	200	5 mm depth from surface of cylinder 2.

Table of Space Ionizing Radiation Exposure Dose Limits (Cont.)

Critical Organ	Average yearly dose (rads)	Maximum permissible single acute emergency exposure (rads)	Location of dose points
Feet, ankles, and hands	175	700	0.07 mm depth from surface of cylinder 3 at highest dose point.
Eyes	25	100	3 mm depth from surface on cylinder 1 along eyeline.

6.2.2.84 Reliability (Equipment). - The probability that an item will perform satisfactorily for a specified period of time when used in the manner and for the purpose intended.

6.2.2.85 Reliability (Test). - The degree to which tests can be made to measure consistently, generally expressed as a coefficient of correlation between test scores obtained on the same subjects at a different time.

6.2.2.86 Reliability Program. - A program established by a system contractor to assure that the reliability requirements fixed by the procuring activity and incorporated in the contract are achieved in the system.

6.2.2.87 Replaceability. - Replaceability (R) refers to those components and parts capable of being replaced or substituted one for the other under field maintenance conditions and without loss of functional operations.

6.2.2.88 Secondary Duty Station. - Areas for taking navigation fixes, performing maintenance, food preparation, and certain scientific observations.

6.2.2.89 Service Propulsion System. - Provides major impulse increments for the Service Module.

6.2.2.90 Service Reaction Control System. - Provides thrust vectors which permit three-axis control and minor thrust increments for minor midcourse velocity corrections of the Spacecraft. This terminology does not include the associated Navigation and Guidance System.

6.2.2.91 Shirtsleeve Environment. - The environmental conditions which permit freedom of movement to crew members while wearing no more than one layer of clothing and being independent of special personal breathing apparatus, cooling or heating garments or other personal environmental control devices.

6.2.2.92 Snorkel. - A system utilizing deployable ports to provide post-landing ventilation on ground and in water.

6.2.2.93 Space Radiator. - The system component that rejects the heat generated within the Spacecraft to the surrounding area by electromagnetic radiation.

6.2.2.94 Super Critical Cryogenics. - Oxygen, nitrogen, and hydrogen stored in the gaseous state at low temperatures and high pressure. Storage pressures are above the critical pressure for the respective gas.

6.2.2.95 Temperature. - In this specification, the temperature to be used in all analyses is the maximum or the minimum temperature calculated for the given loading condition, whichever is more critical.

6.2.2.96 Thermal Control. - Temperature control of the electrical, electronic and mechanical components aboard the Spacecraft.

6.2.2.97 Toilet. - A structure into which crew members can void feces.

6.2.2.98 TVC. - Thrust vector control.

6.2.2.99 UHF. - Ultra high frequency, 300-3000 mc.

6.2.2.100 Ultimate Factor of Safety. - The ratio of the design ultimate load on a structure to the limit load.

6.2.2.101 VHF. - Very-high frequency, 30-300 mc.

6.2.2.102 Wall Area. -That vertical area between the floor and ceiling area which begins at the intersection of floor and vertical wall and ends at the intersection of vertical wall and sloping wall or ceiling.

6.2.2.103 Warning Light. -A placard light used to inform the crew members of the existence or occurrence of a hazardous condition requiring immediate corrective action, such as fire warning or cabin pressure failures.

6.2.2.104 Yield Factor of Safety. -The ratio of the design yield load on a structure to the limit load.